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LFC Leading Edge Glove Flight - Aircraft Modification Design, Test Article Development, and Systems Integration

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CONTRACT NAS1-16219
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National Aeronautics and Space Administration

Langley Research Center
Hampton, Virginia 23665

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FOREWORD

Contract NAS1-16219 between the National Aeronautics and Space Administration and the Lockheed-Georgia Company, effective September 8, 1980, provides for the test article development and aircraft modification design activities necessary for the integration of leading-edge test articles into the NASA JetStar aircraft. The contract is sponsored by the Aircraft Energy Efficiency Project Office of the Langley Research Center, with M. C. Fischer serving as Technical Monitor. This document, submitted in fulfillment of DRL-I-007 of the subject contract, constitutes the final report.

The International System of Units (SI) are presented as primary, and customary units are in parentheses. Customary units were used for the principal measurements and calculations. Report No. NAS SP-7012, "SI Units, Physical Constants and Conversion Factors," 2nd Revision, E.A. Mechtly, was used as the basis for conversion.

At the Lockheed-Georgia Company, the contract was accomplished under the cognizance of R. H. Lange, Manager, Advanced Concepts Department, with F. R. Etchberger serving as Project Manager. Principal participants in this contract effort were as follows:

J. A. Bennett	Aerodynamics	R. S. Ferrill	LFC Systems
L. B. Brandt	Aerodynamics	J. G. Tibbetts	LFC Systems
J. R. Sanders	Design	L. B. Lineberger	Structures
H. J. Abbey	Design	P. L. Kesler	Structures
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G. P. Dearing	Design		

This final report was compiled and edited by C. A. Simica who replaced F. Etchberger as Project Manager upon Mr. Etchberger's retirement on 1 April 1983.

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ABBREVIATIONS

ACEE	Aircraft Energy Efficiency
AI	anti-icing
CA	contamination avoidance
CAR	Civil Aeronautics Regulation
CEBECI	laminar boundary layer analysis computer code
CRT	cathode ray tube
DAC	Douglas Aircraft Company (McDonnell Douglas)
deg	degrees (angular)
DFRF	Dryden Flight Research Facility
EGBE	ethylene glycol monobutyl ether
EP	emergency pressurization
FL022NM	transonic, full potential, wing alone analysis with 2-D strip boundary layer - computer code
FS	fuselage station
FWD	forward
HF	hot-film gauge
HYD	hydraulic
ID	inside diameter, cm (in)
IP	ice protection
KEAS	knots equivalent airspeed, m/sec (ft/sec)
KTAS	knots true airspeed, m/sec (ft/sec)
LaRC	Langley Research Center, Hampton, VA
L.E., LE	leading edge
LEFT	leading-edge flight test
LES	leading edge section

LETA	leading-edge test article
LFC	laminar flow control
MAC	mean aerodynamic chord, m(ft)
MTC	Mach trim compensator
NASA	National Aeronautics and Space Administration
NORAIR	former division of Northrop Corporation
O/B, OB	overboard
OD	outside diameter, cm(in)
PCM	pulse code modulation
PGME	propylene glycol methyl ether
rad	radians
SALLY	boundary layer stability analysis computer code
SOV	shut-off valve
T.E., TE	trailing edge
TOFL	takeoff field length
TS	Tollmien-Schlichting
TSP	turbocompressor suction pump
TYP	typical
WL, W/L	water line
WRP	wing reference plane
WS	wing station

SYMBOLS

C	cleaning slot
C_D	drag coefficient
C_L	lift coefficient
C_P	pressure coefficient
C_{PS}	slot pressure coefficient
C_q	suction coefficient
D	dual-purpose slot
dB	decibels
Hz	Hertz (frequency in cycles per second)
KPa	kilopascals
L	lower slot
M	Mach number; moment (bending) load, M-N/M (in-lb/in)
M_D	dive Mach number
N	revolutions per minute
N factor	natural logarithm of the ratio of a boundary layer disturbance amplitude to its amplitude at neutral stability
P	tension (+) or compression (-) load, N (lb)
q	dynamic pressure, KPa (lb/sq in)
R, Rn	Reynolds number
Rw	slot width Reynolds number
t/c	airfoil thickness to chord ratio
U	upper slot
Uz/Ue	velocity at sucked height of boundry layer/ velocity at edge of boundary layer
V	shear load N(lb)

V_D	design dive speed, m/sec, (ft/sec)
W/Z	slot width/sucked height of boundary layer
Wcorr	corrected airflow, kg/sec(lb/sec)
X-21	Northrop LFC test aircraft (B-66)
x/c	"x" distance from leading edge to chord ratio
α	angle of attack, rad (deg)
δ_F	flap deflection angle, rad (deg)
ϵ-N	strain - cycles to failure
η	wing semispan location ratio (WS/semispan length)
θ	ambient temperature/standard day temperature correction factor
∞	slope of ϵ-N curve

1.0 SUMMARY

This report documents Lockheed's results in designing, testing, and fabricating a laminar flow control (LFC) leading-edge flight test article and associated subsystems for the NASA. McDonnell Douglas, in a separate contractural effort with NASA, fabricated a porous surface LFC leading-edge flight test article. Lockheed performed as the integration contractor for both articles. This report addresses only the Lockheed flight test article and aircraft modifications up to the McDonnell Douglas interface.

The overall objective of NASA Contract NAS1-16219 is to provide operationally effective leading-edge systems for NASA flight testing under flight conditions representative of future commercial transport aircraft operations. A continuing Lockheed objective is to support the NASA during JetStar functional testing and subsequent LFC data collection and analysis.

The Lockheed designed LFC leading-edge flight test article is comprised of a fiberglass/epoxy substructure incorporating slot ducts, metering holes, and collector ducts covered by a titanium skin in which a total of 27 slots, 0.0094 cm (0.0037 in.) in width, are cut in the upper and lower surfaces. Also incorporated in the leading-edge are the required suction plumbing, cleaning and purge system plumbing, and pressure recording instrumentation. The article fits the JetStar left-hand wing leading edge between wing stations 134.75 and 196.00 and aft to the front spar at the 12 percent chord location. The McDonnell Douglas flight test article fits the right-hand wing at the same wing stations and also attaches to the front spar. A Garrett Corporation (AiResearch) turbocompressor suction pump (TSP) installed in the area aft of the fuselage pressure vessel will provide the suction source necessary to operate both LFC leading-edge systems. Associated with the TSP are three chamber valves, mounted in the passenger compartment, used to control suction pressures for the individual test articles. Also within the passenger compartment are the LFC systems operator's consoles and instrumentation and data collection subsystems.

To prevent contamination of the leading-edge flight test articles due to insect accretion during takeoff and landing operations, each flight test article incorporates a cleaning system. The Lockheed article uses two dedicated cleaning and six dual-purpose slots through which a cleaning/anti-icing fluid flows onto the surface. The McDonnell Douglas article deploys a shield from the lower leading-edge surface to shield the leading-edge and to allow operation of spray nozzles. To clear the cleaning system a primary purge air system is available for operation at altitudes above 3,658 m (12,000 ft). A secondary system provides purge air from the aircraft air-conditioning system. A nitrogen pressurization system is used to provide pressurization for the liquid (cleaning/anti-icing fluid) reservoirs of both the Lockheed and McDonnell Douglas liquid systems, McDonnell Douglas cleaning system purge, and instrumentation purge.

To incorporate all the required LFC systems, modifications to the NASA JetStar were required. Lockheed designed the modifications and fabricated many of the parts required to complete this modification. The JetStar wing slipper fuel tanks were removed to provide the location of the LFC flight test articles. The inboard leading-edge flap section on each wing is also removed for the same reason. The two outboard leading-edge flap sections are locked in the retracted

position for the duration of the LFC flight test program. A trailing-edge flap section was fabricated to fill the gap in the wing trailing edge created by removal of the wing fuel tanks. This trailing-edge flap section on each wing also now contains the landing lights previously mounted in wing fuel tanks. De-icer boots are installed on the wing fixed leading-edge sections and on the two outboard leading-edge flap sections. Modification designs were also completed to enable suction tubing and instrumentation routing from the wings through the fuselage skins to the chamber valves and the control consoles. The fuel quantity gauging system and refueling system were also modified. Due to the fuel tank removals, the fuel capacity of the modified JetStar is reduced from 8,084 kg (17,822 lb.) to 4,383 kg (9,662 lb.) or by 3,700 kg (8,160 lb.).

Due to the structural modifications to the JetStar and LFC systems incorporation, Lockheed provided ground acceptance and flight acceptance procedures to the Dryden Flight Research Facility for their implementation. Incorporated in these procedures are provisions for a ground vibration survey and a flight flutter test. The flight flutter tests are scheduled subsequent to a flight readiness review to be conducted in September/October 1983.

Airfoil shape and suction system requirements were determined through extensive wind-tunnel testing and computer model simulations. These aerodynamic tests and flow analyses defined the leading-edge contour, suction airflow values, slot widths, slot duct shape, metering hole sizes, collector duct size, and suction line sizes. These design criteria were translated to a set of engineering drawings that were used by manufacturing personnel to fabricate the unique LFC flight test article.

Fabrication of the LFC test articles was characterized by trial and error methods since an article of this type had never been fabricated. Fabrication of the substructure was a step-by-step operation from positioning of slot duct inserts, bonding of collector ducts and honeycomb core, to layup and cure of the inner skin. To form the outer surface, two layers of titanium bonded together was tried; however, this proved unacceptable due to slot closure and flattening. Hot-forming the titanium skin was tried but, again, was unsuccessful due to skin "oil canning." Finally, a technique of roll forming followed by stress relieving proved successful in forming the titanium skin for the flight-test article. Slotting the titanium skin was another operation for which different techniques were tried and discarded. A procedure was developed using a computer controlled mill and jewelers saws to complete the skin slotting operation. Cutting of these precision slots in the titanium skin was a major accomplishment of the fabrication program.

A sonic fatigue and ultimate strength test article was fabricated using the manufacturing techniques that evolved during the program. The two test articles were fabricated as one piece and cut in half to produce the two separate test articles. The titanium skin was hot formed for these test articles; the flight test article skin was roll formed.

Although flow of adhesive FM123-4 had not occurred in titanium skin to substructure bonding operations prior to fabrication of the flight-test article, flow of this adhesive into portions of the slot ducts in the flight-test article almost damaged this article beyond repair. The aft four upper and three lower

surface suction slots were effectively blocked by adhesive. The other slots had varying degrees of adhesive blockage. A repair technique was proved on the expendable sonic fatigue article. The aft blocked slots were repaired by cutting the titanium skin in strips the length of the slots, filling and recutting the slot ducts, and bonding hand-fitted titanium skin strips in place. The adhesive flow characteristic was eliminated by pre-heating the adhesive prior to bonding the repair skin strip. Repair of the remaining slot ducts was accomplished by cutting a slot at each end of the slot ducts to enable a thin-walled tube to be powered slowly along the slot duct. Suction flow was significantly recovered on the flight-test article using these repair techniques. Subsequent flow measurements and computer analysis deemed the article acceptable for flight test. The LFC leading-edge flight-test article was Government accepted on April 25, 1983 and arrived at the Dryden Flight Research Facility on April 26, 1983.

Design and fabrication of the LFC flight-test article, design of the aircraft modifications, and integration of the associated LFC subsystems and the McDonnell Douglas flight-test article into the JetStar aircraft, represent significant accomplishments in advancing LFC technology towards application to commercial transport aircraft and concomitant improved fuel efficiency.

2.0 INTRODUCTION

The recognition of potential long-term shortages of petroleum-based fuel, evidenced by dramatic increases in costs and periods of limited availability since 1973, emphasized the need for improving the fuel efficiency of long-range transport aircraft. In 1976, in response to this need, the NASA established the Aircraft Energy Efficiency (ACEE) program with the objective of maintaining the U. S. competitive advantage through development of new technology for fuel efficiency. Of all advanced-technology concepts currently under consideration for application during the next two decades, Laminar Flow Control (LFC) offers the greatest potential for improving the fuel efficiency of transport aircraft. Consequently, LFC is included as one element of the ACEE program, and the NASA formulated a three-phase program with the goal of developing LFC technology to permit application to aircraft in the 1990 period.

The Phase I effort, concluded in September 1978, resulted in the definition of candidate LFC systems for application to future production aircraft. Phase II, of which this contract is a part, involves initial development and testing of selected leading-edge subsystems, and the design and development of selected structural concepts. The final phase, Phase III, originally envisioned to encompass the design, fabrication, and flight demonstration of an integrated LFC system in a validator aircraft, will be redefined at some future time.

This leading-edge flight test program will evaluate the effectiveness of LFC systems under representative flight conditions. Operable LFC leading-edge systems -- suction, cleaning, and anti-icing -- will be installed on the NASA JetStar aircraft. Two contractors, Lockheed and McDonnell Douglas, will each provide a leading-edge test section for simultaneous testing.

This report covers the work accomplished by Lockheed (integration contractor) to design and fabricate a leading-edge test section and fairings, to design aircraft structural modifications and LFC systems installations, and to define ground and flight acceptance test procedures. This report covers these efforts from contract go-ahead, September 8, 1980, through delivery of the Lockheed test article, April 25, 1983.

3.0 OBJECTIVES

The overall objective of NASA Contract NAS1-16219 is to provide operationally effective LFC leading-edge systems for flight testing under flight conditions representative of future LFC commercial transport aircraft operations. LFC leading-edge test articles were fabricated to demonstrate that the required LFC systems can be packaged into a leading-edge section of a wing representative of future LFC commercial transport aircraft. A functional checkout of these LFC leading-edge systems will be achieved in flight.

Specific objectives for Lockheed (integration contractor) are to

(a) Provide an operable LFC leading-edge system for the leading-edge section of the test aircraft and wing fairings for both sides of the aircraft.

(b) Assure the leading-edge systems can be safely flown on the test aircraft (NASA JetStar).

(c) Recommend any aircraft operational procedure changes.

(d) Provide the test aircraft modification design, and provide off-site support during aircraft modification and installation of test articles.

(e) Provide support of systems checkout to demonstrate operations of LFC systems and to instruct NASA personnel in system operations.

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4.0 LEADING EDGE DESIGN AND FABRICATION

4.1 DESIGN

One of the specific objectives of the LFC-LEFT program is to provide operable LFC test articles for the wings of the NASA JetStar with fairings for both wings. A McDonnell Douglas designed test article is to be installed on the opposite wing from the Lockheed article. Figure 1 shows the JetStar with the test articles installed. Test articles are positioned on the left and right wings in the area of the wings where the JetStar external fuel tanks are normally installed. The external tanks are removed to provide adequate space for the test articles and necessary fairings. This space is readily adaptable to this use, as the wing leading and trailing edges are not continuous under the external tanks and their fairings.

A plan view of the JetStar wing, Figure 2, shows the basic installation geometry of the Lockheed article on the left wing and the McDonnell Douglas article on the right wing. Externally, the sections are symmetrical, the only apparent difference being that the McDonnell Douglas sensor panel is larger chordwise than the Lockheed sensor panel.

As shown, the portion of the wing revised for test extends from wing station 122.068 to wing station 205.278 and extends from the wing leading edge over the top of the wing to 65 percent chord of the basic JetStar wing. The gloved test section also extends from the wing leading edge under the wing to approximately 14 percent chord on the lower surface.

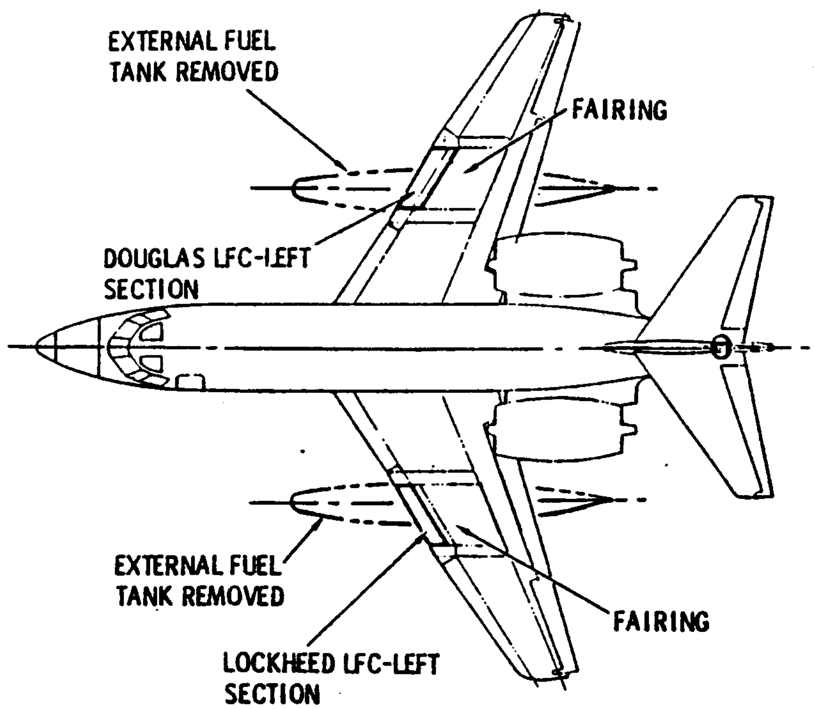

Figure 1. LFC-LEFT JetStar

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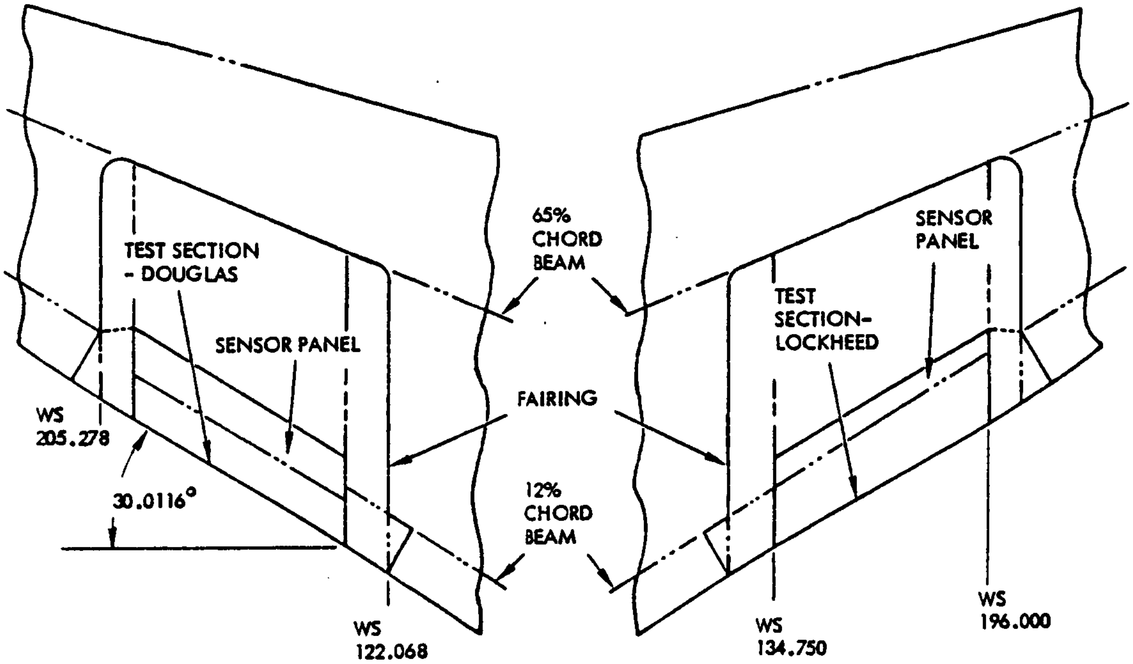

Figure 2. Wing Planform

Airfoil sections jointly developed by NASA, Lockheed, and McDonnell Douglas were applied to the JetStar wing at wing stations 134.750 and 196.000. The test articles and fairings were developed around these control stations. The control airfoils were designated MOD8c by Lockheed. Transition fairings were developed between wing stations 122.068 and 134.750 and between wing stations 196.000 and 205.278 to bring the glove section surfaces back to basic JetStar wing contours.

Figure 3 shows the leading edge sweeps of the basic JetStar wing and the leading edge test article.

4.1.1 Test Article Assembly

The leading-edge test article is a complete, shippable assembly. The assembly shown in Figure 4 is designed to be supported on the JetStar wing by upper and lower beam cap extensions installed on the front beam. Active slot area is 119.23 cm (46.94 in) wide, which extends from wing station 141.140 to wing station 188.080.

There are 27 slots -- 12 in the upper surface and 15 in the lower surface. Only 26 suction lines are provided for connection to the LFC aircraft system installations, as 2 of the forward-most slots (designated for cleaning only) are combined by internal plumbing. All suction tubing, instrumentation tubing, and hot-film instrumentation are installed as an integral part of the assembly.

Major structural assemblies of the test section are the leading-edge surface panel, forward diaphragm, aft diaphragm, diaphragm support fitting, diaghragm support angles, and a rib on each end.

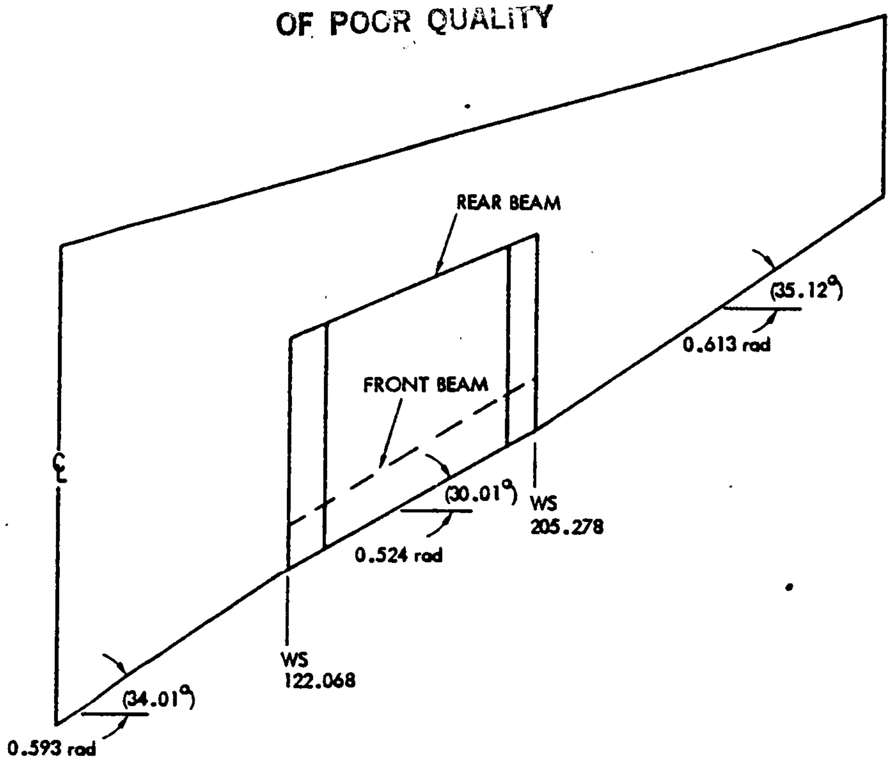

Figure 3. Leading Edge Sweep

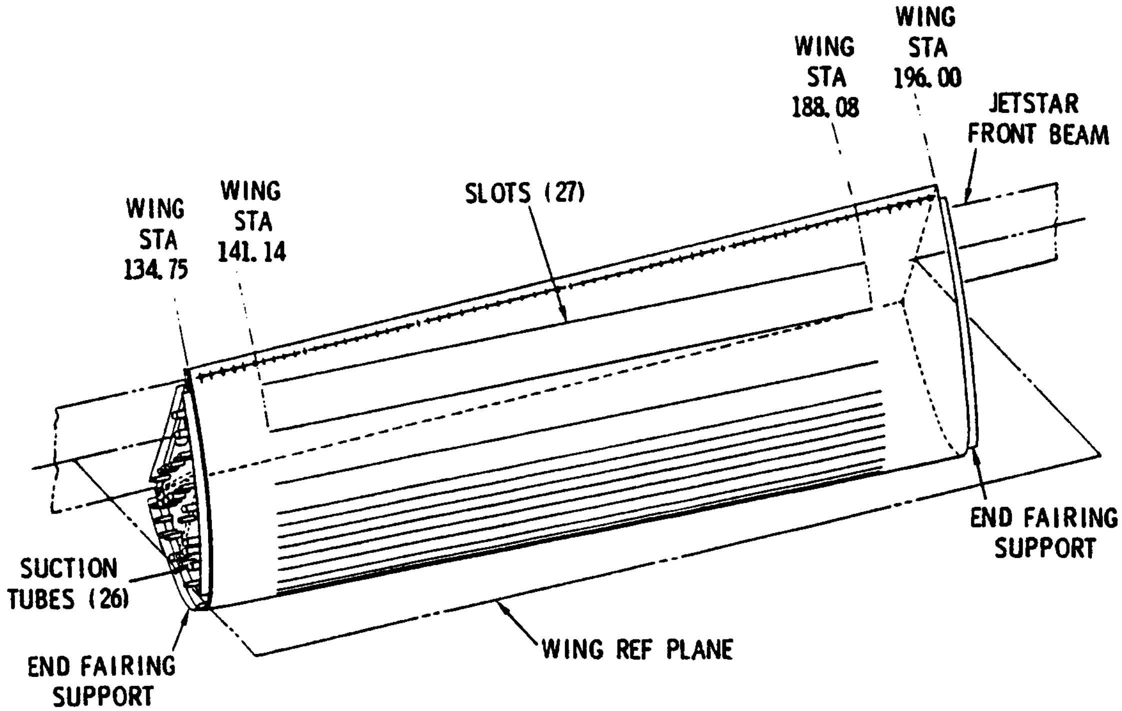

Figure 4. Test Article Assembly

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4.1.1.1 Surface Panel

The test article surface panel, shown in Figure 5, is a composite shell of graphite and fiberglass with a titanium outer skin. The panel is 155.58 cm (61.25 in) long, 31.12 cm (12.25 in) high at the inboard end and 29.16 cm (11.52 in) high at the outboard end. The 49.83 cm (19.62 in) chord at the outboard end is longer than the 41.99 cm (16.53 in) chord at the inboard end. The longer chord at the outboard end is the result of the reduction of the test section leading-edge sweep from that of the basic JetStar wing.

Figure 6 shows the composite build-up of the test article surface panel. Components of the panel are the inner and outer face sheets, slot ducts, collector ducts, core materials, outer skin, and the doublers and filler required. A Lockheed-Georgia process specification was written to control and outline the process of assembling the leading-edge test article.

The inner and outer face sheets are composed of five plies of 0.02 cm (0.008 in) graphite-epoxy prepreg fabric, "Fiberite" HMF 343/34. The outer face sheet is contoured to fit the slot ducts. The slot ducts are molded strips made from fiberglass prepreg fabric. The collector ducts are extruded sections fabricated of E-Glass roving and vinyl-ester resin. The core material is nylon (NOMEX). Adhesive, AF 3015 or FM 37, is used to bond the core to the collector duct and as a filler where the collector ducts are too close together to allow installation of any meaningful piece of core. A single sheet of 0.05 cm (o.02 in) 6AL-4VTI Type IIIC titanium per MIL-T-9046H is cold-formed and then stress-relieved to make the outer skin. After forming, the skin is cleaned and etched

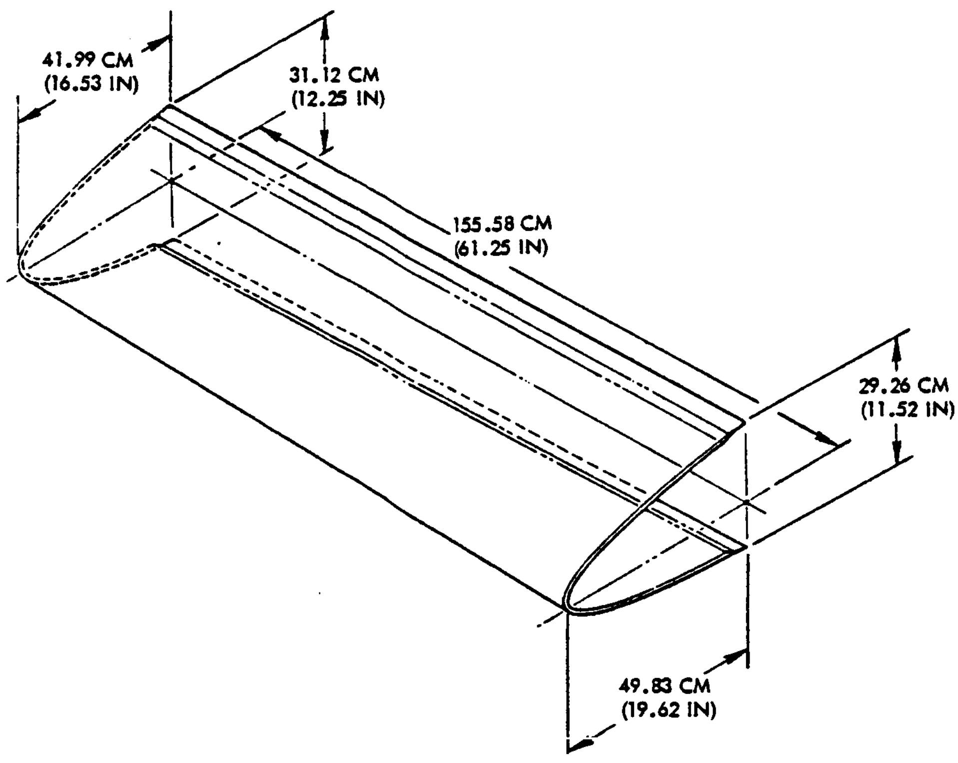

Figure 5. Surface Panel

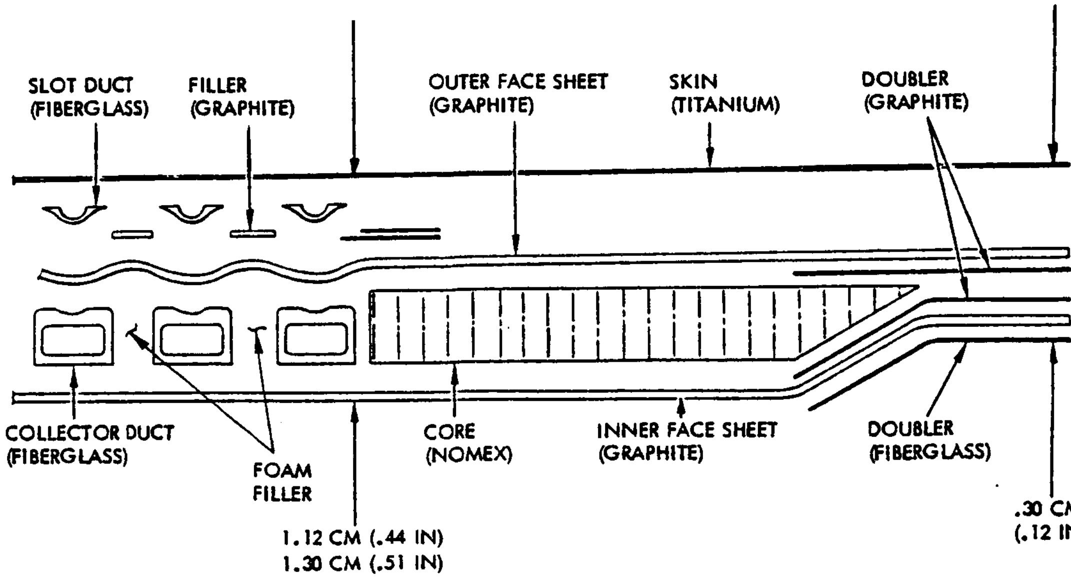

Figure 6. Composite Build-Up

to bring the thickness to 0.041 cm (0.016 in). Graphite-epoxy and fiberglass doublers and fillers are added in the test article shell as required.

Adhesives per FM 400 film per FED Spec MMM-A-132 Type II and MIL-A-25463 Type II Class 1 are used to bond the face sheets to the core materials, and adhesive FM 123-4 per FED Spec MMM-A-132 Type I, Class 2 is used to bond the titanium skin to the composite test section.

The aft edge of the test article shell is reduced to 0.30 cm (0.12 in) thickness to facilitate attachment of the test section to the JetStar wing.

A spanwise phenolic strip per MIL-P-15035 type FBM is installed in place of the NOMEX core under the test article diaphragm attach points for reinforcement. The ends of the test article shell are potted using STM 22-602 Type V Grade B Class 1 material. This construction provides support for the end rib caps and closes the ends of the collector ducts. Core potting is also done using EA 934 adhesive per FED Spec MMM-132 Type I Class 3 material.

Figure 7 shows a typical cross section of the test article. Relative slot location and slot identification are shown. Slots U1 thru U11 are uppersurface, suction-only slots. Slots L1 thru L8 are lower-surface, suction-only slots, Slots D1 thru D6 are dual-purpose cleaning/suction slots, and slots C1 and C2 are cleaning-only slots. Control of the slot location is applied at the inboard end of the active slot, wing station 141.140. The "x" dimension, Figure 7, along with the slot width is shown in Table 1. Slots are cut parallel to the leading edge of the test article.

4.1.1.2 Diaphragms

Two diaphragms are installed in the test article to constrain deflection of the surfaces when air loads are applied. These structural members run spanwise

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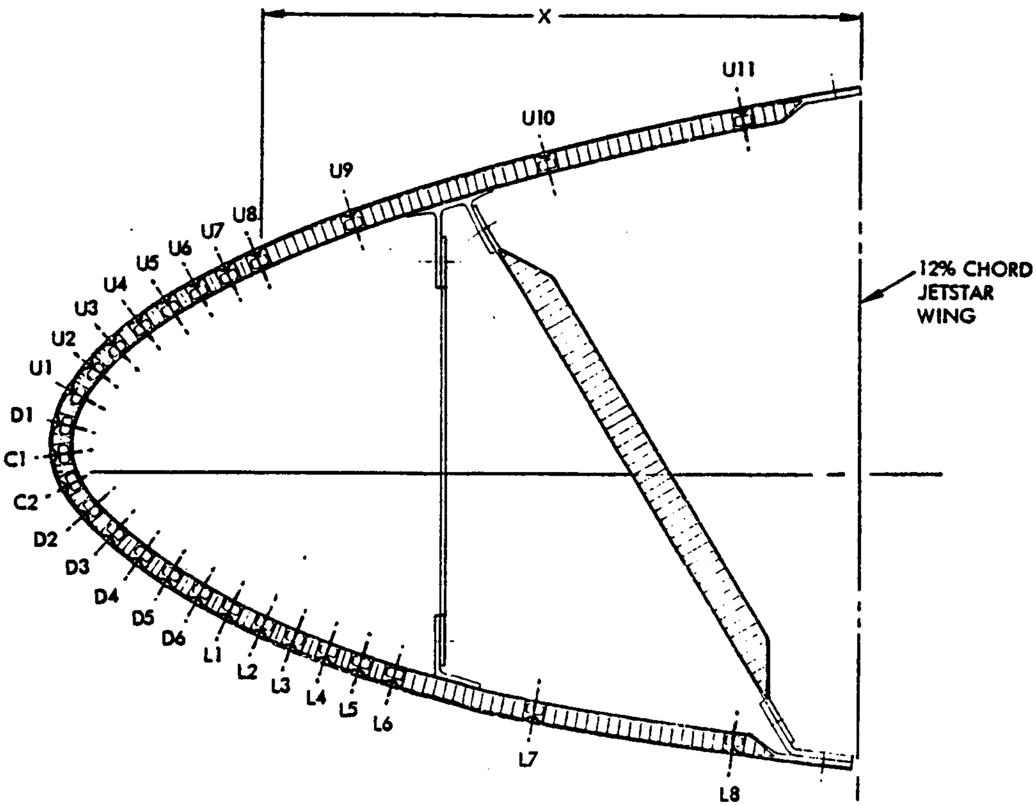

Figure 7. Test Article Cross-Section

across the full length of the test article. The lower edge of both the forward and aft diaphragm attaches to the test section shell through stainless-steel angles bonded and bolted to the shell. The upper edges attach to a common composite fitting, which is also bonded and bolted to the shell. Adhesive FM 123-2 per FED Spec MMM-A-132 Type I Class 2 is used for the bonding material. The steel angles are 301 1/4 hard stainless per MIL-S-5059. Figure 8 shows the upper diaphragm attach fitting, which is made up of graphite-epoxy prepreg fabric "Fiberite" with lower surface face sheets of fiberglass. Nutplates are installed on the angles and composite fitting to facilitate attachment of the diaphragms.

Forward Diaphragm

The forward diaphragm is made in five sections to facilitate the installation of suction and instrumentation tubing. Figure 9 shows the forward diaphragm installation. The web sections are made from 0.318 cm (0.125 in) thick 2024T3 clad aluminum plate material. Web plates are bolted together through stiffeners of 7075-T6511 aluminum bulbed "T" sections. Lightening holes are provided in the beam web to facilitate installation of suction tubing. Space is provided along the web for installation of suction tube supports. The ends of the diaphragm are tied to the test article end ribs through bent-up sheet angles made from 301 1/4 hard stainless steel per MIL-S-5059.

TABLE 1. LOCKHEED SLOT GEOMETRY

SLOT GEOMETRY

UPPER SURFACE SLOT IDENT.	LOCATION* "X" CM	LOCATION* "X" IN	SLOT** WIDTH CM	SLOT** WIDTH IN
D1	42.621	16.780	0.0094	0.0037
U1	41.933	16.509	0.0094	0.0037
U2	40.917	16.109	0.0094	0.0037
U3	39.685	15.624	0.0094	0.0037
U4	38.298	15.078	0.0094	0.0037
U5	36.817	14.495	0.0094	0.0037
U6	35.270	13.886	0.0094	0.0037
U7	33.685	13.262	0.0094	0.0037
U8	32.073	12.627	0.0094	0.0037
U9	26.850	10.571	0.0102	0.0040
U10	16.688	6.570	0.0114	0.0045
U11	6.403	2.521	0.0127	0.0050
LOWER SURFACE				
C1	42.258	16.637	0.0094	0.0037
C2	42.187	16.609	0.0094	0.0037
D2	41.138	16.196	0.0094	0.0037
D3	39.807	15.672	0.0094	0.0037
D4	38.341	15.095	0.0094	0.0037
D5	36.792	14.485	0.0094	0.0037
D6	35.187	13.853	0.0094	0.0037
L1	33.541	13.205	0.0094	0.0037
L2	31.864	12.545	0.0094	0.0037
L3	30.155	11.872	0.0094	0.0037
L4	28.438	11.196	0.0094	0.0037
L5	26.657	10.495	0.0094	0.0037
L6	24.882	9.796	0.0094	0.0037
L7	17.280	6.803	0.0102	0.0040
L8	8.293	3.265	0.0114	0.0045

* $\pm$0.0381 CM (0.0150 IN)

** $\pm$0.0013 CM (0.0005 IN)

Aft Diaphragm

The aft diaphragm is a composite structure built with a twist in the surface. The twisted or warped diaphragm surface is necessary to allow the upper end of the diaphragm to intercept the test article shell between slot locations along its full span and allow the lower edge of the diaphragm to properly match the lower cap of the JetStar wing front beam to avoid a cumbersome load path to the beam cap. Figure 10 shows the twist built into the aft diaphragm. Figure 11 shows the build-up of the diaphragm. The face sheets doublers are made from graphite-epoxy prepreg fabric "Fiberite," HMF 133/34 and HMF 343/34, respectively. The core is nylon NOMEX. Bonding of the assembly is accomplished using FM 400 adhesive. Top and bottom edges of the diaphragm are reduced in thickness to facilitate diaphragm fitting with the support fitting and angle. The ends of the diaphragm are potted.

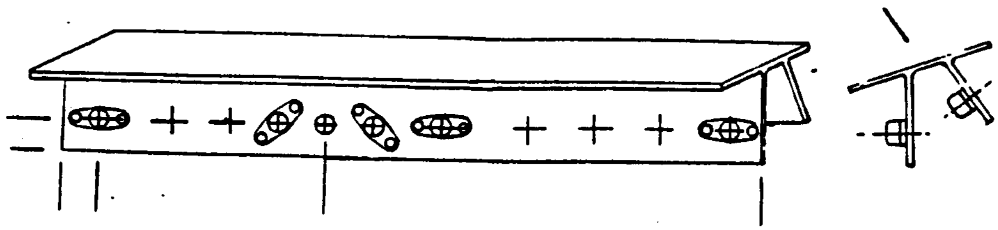

Figure 8. Diaphragm Attach Fitting

Aft diaphragm installation is shown in Figure 12. The upper edge is fastened to the upper fitting common to the forward diaphragm and the lower end to the attach angle which attaches to the JetStar wing. Ends of the diaphragm are tied to the test article end ribs with stainless-steel angles similar to those typing the ends of the forward diaphragm. Space is provided five places along the diaphragm for suction-tube supports.

4.1.1.3 End Ribs

Ribs are installed on each end of the test article with screws and nutplates. Figure 13 shows the outboard rib installation, which consists of a web and two bent-up sheet cap angles. As shown in the figure, one cap turns inboard and attaches to the test article shell, and one turns outboard for end fairing attachment. Holes in the web are for suction tubes. The inboard cap is made from 302 CRES AMS 5515 ANL and the outboard cap is made from 2024-T0 clad aluminum and heat-treated after forming to T42. The web is 2024-T3 clad aluminum.

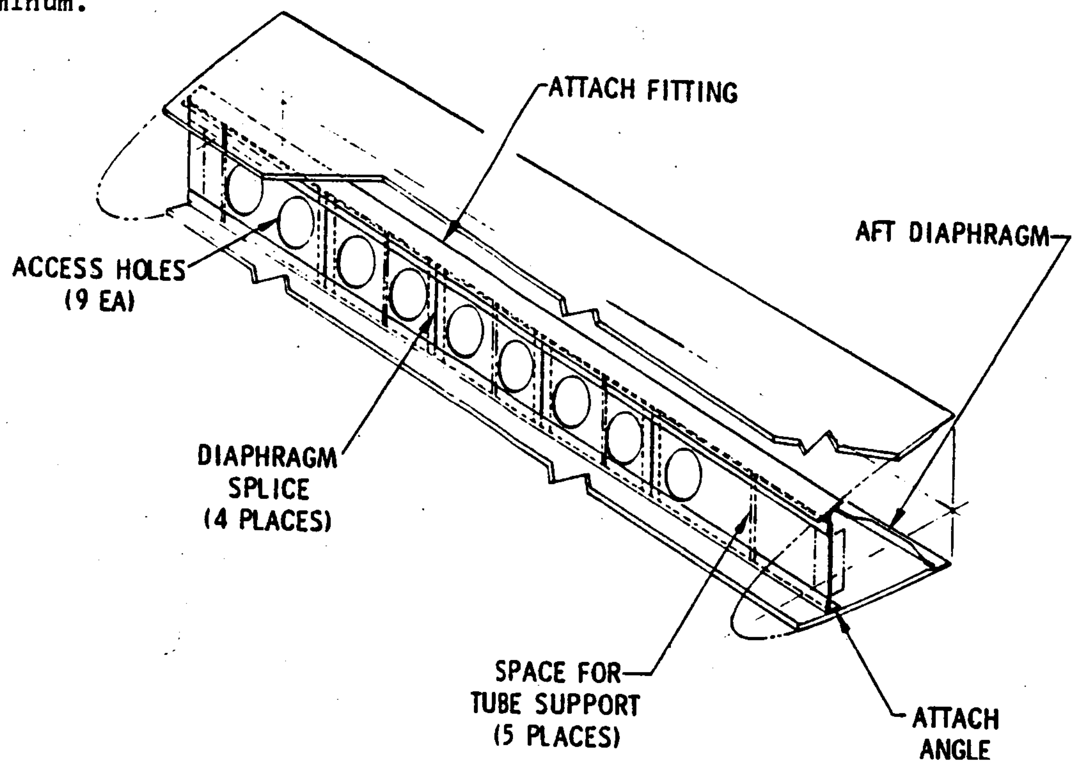

Figure 9. Forward Diaphragm Installation

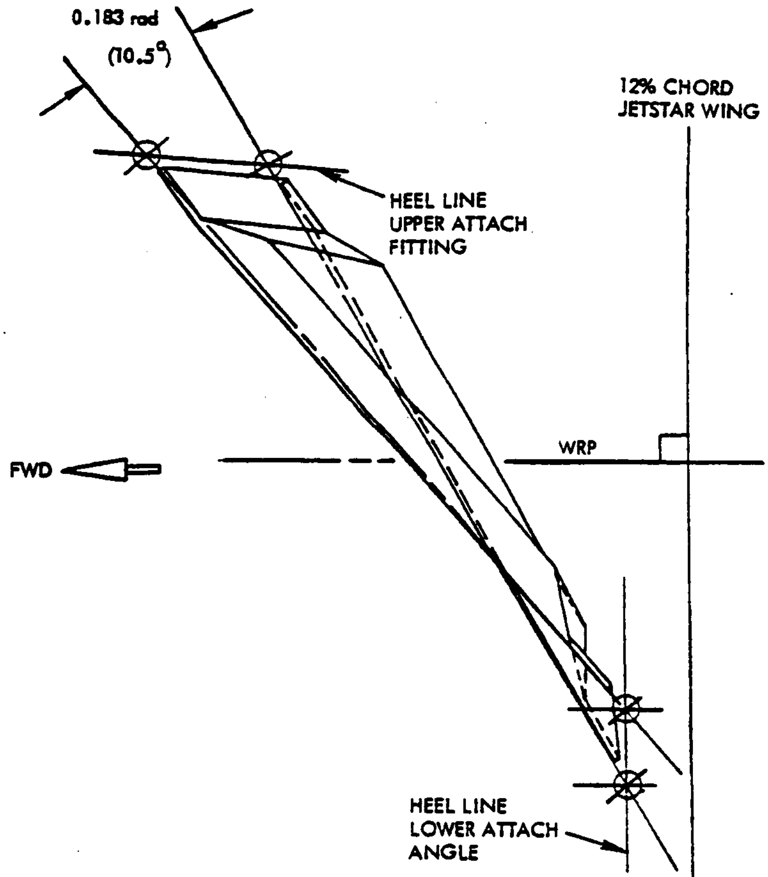

Figure 10. Aft Diaphragm Twist

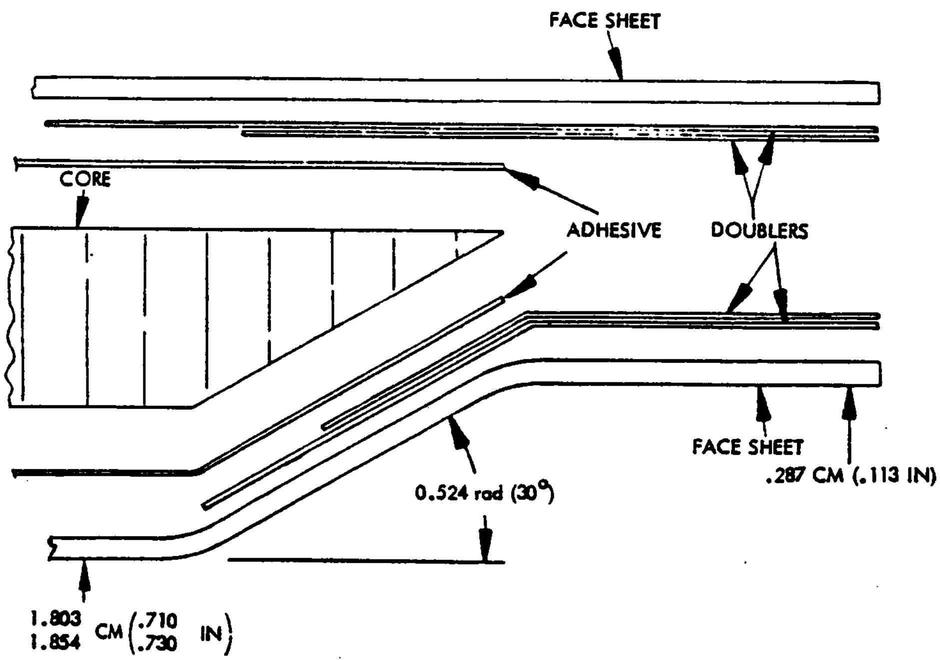

Figure 11. Aft Diaphragm Build-Up

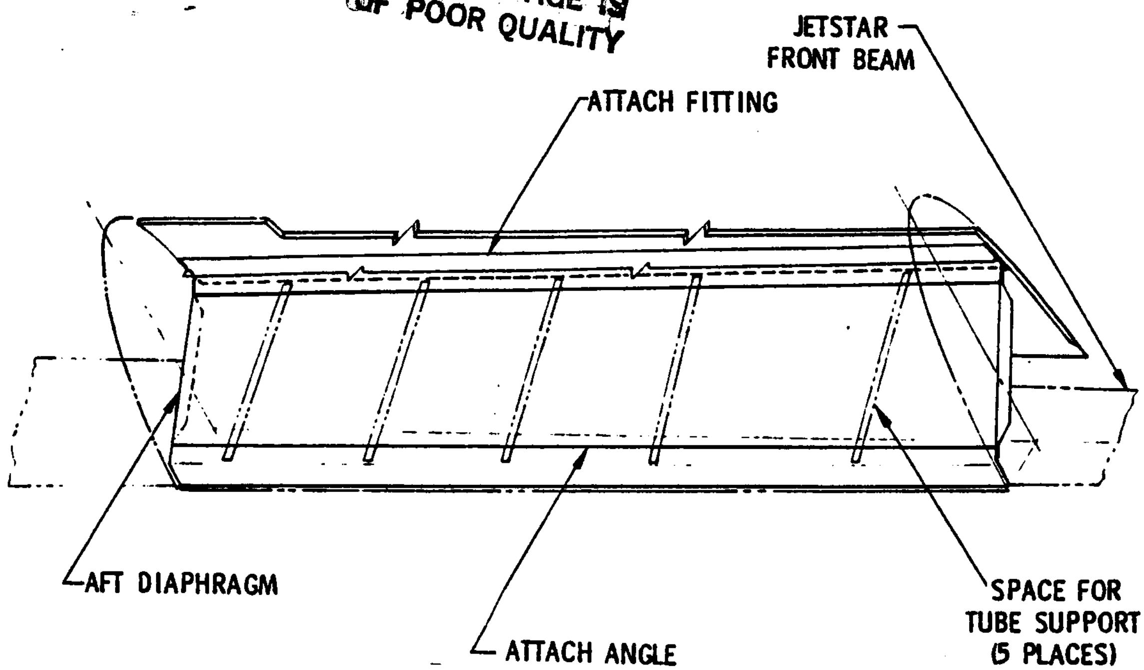

Figure 12. Aft Diaphragm Installation

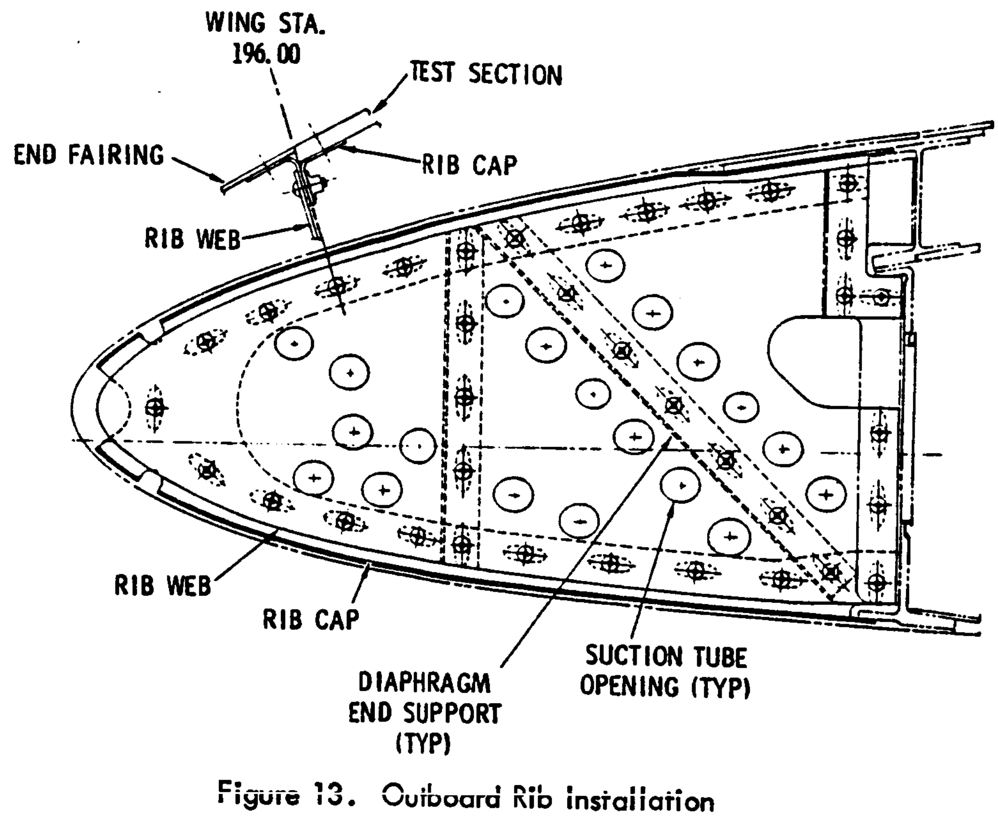

Figure 13. Outboard Rib Installation

4.1.2 Suction Provisions

Suction is applied to all 27 slots. Typical provision for suction in the test article shell is shown in Figure 14. Those slots that are also used as cleaning liquid slots, have a dam installed every 15.25 cm (6.00 in) in the collector duct to facilitate liquid distribution along the leading edge span.

Air enters through the slot into the slot duct and passes through metering holes into the collector duct. Air passes through the collector duct outlets into the test article internal plumbing lines. Metering holes are 0.076 cm (0.030 in) in diameter for all slots, except U11, which are 0.089 cm (0.035 in) in diameter, and are spaced along the active slot at 0.508 cm (0.200 in) intervals. The collector duct outlets are 0.478 cm (0.188 in) in diameter and are spaced at 15.24 cm (6.00 in) intervals along the duct.

Figure 15 shows the surface suction attachment arrangement installed on the inner surface of the test article shell. Some tubes are bonded directly to the surface as shown in Figure 16; other slots have suction connections protruding from the surface. Tubes bonded directly to the surface are generally referred to as manifold tubes. Either space provisions or the liquid-carrying requirement determined which type of suction connection was used on any one slot. Surface tubes are made from 2024-T3 or 6061-T6 aluminum.

Figure 17 shows a cross-section of a typical surface-bonded tube installation. The tube support is fiberglass and is trimmed from a portion of the

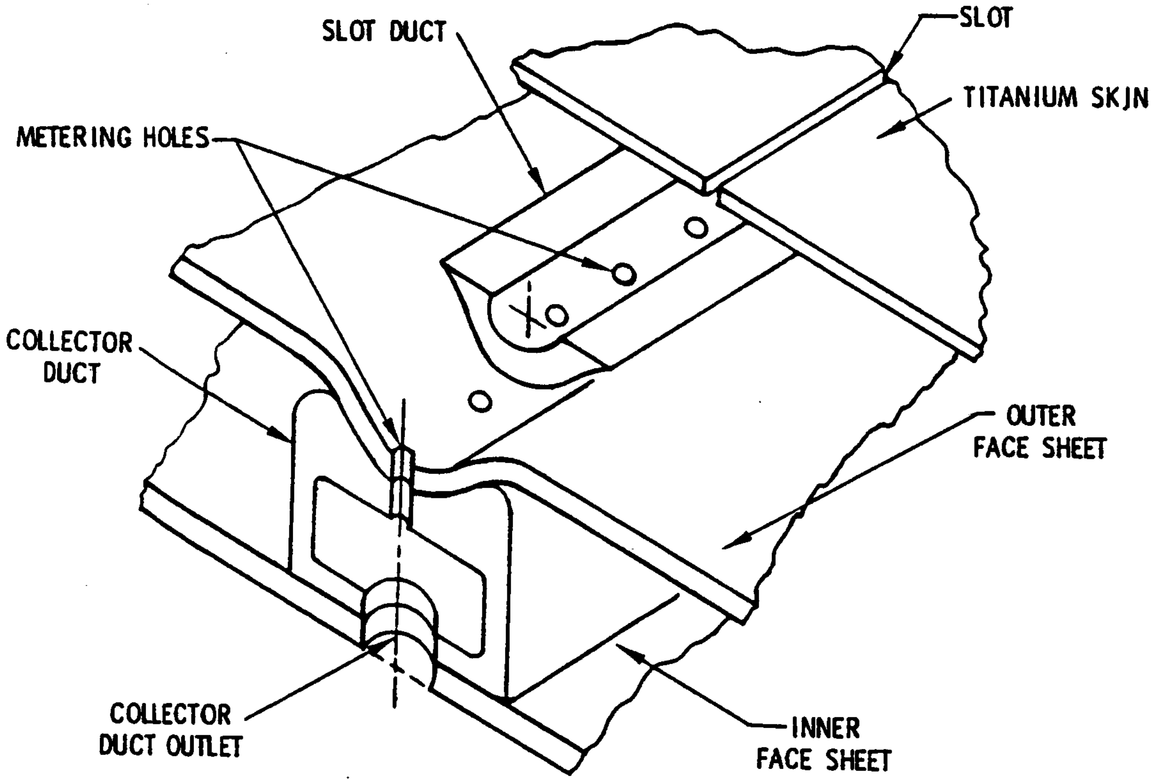

Figure 14. Surface Suction

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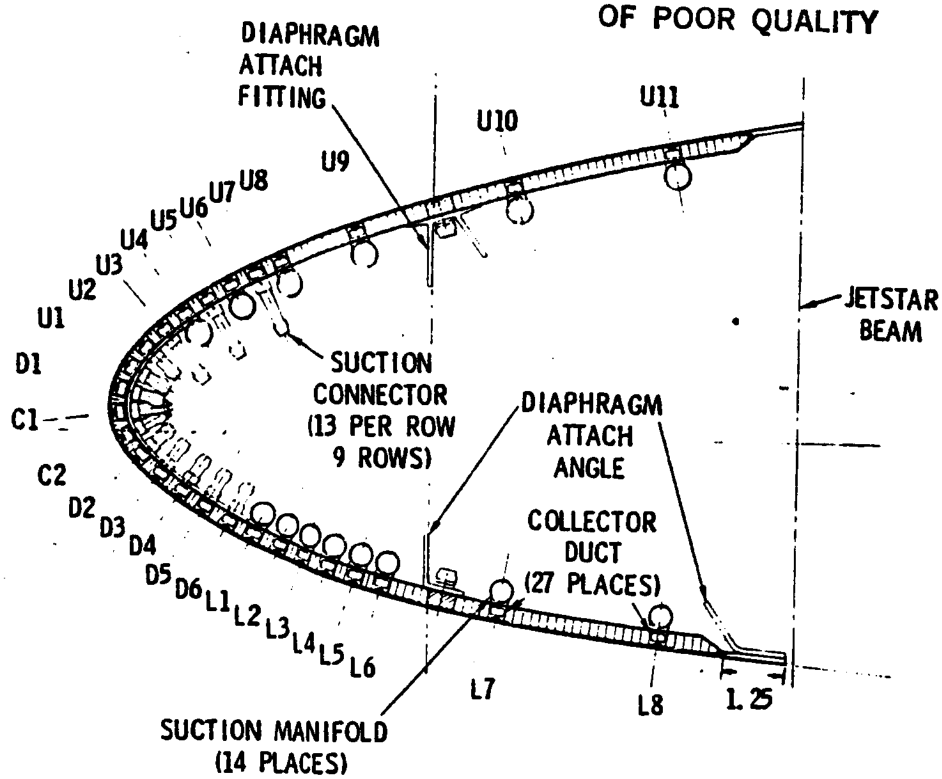

Figure 15. Surface Suction Attachment

collector duct extrusion. Tube to tube-support and tube-support to leading edge surface are bonded with FM 123-2. To assure proper alignment, the collector duct-to-suction tube opening, 0.478 cm (0.188 in), is drilled from the back side of the tube. After the hole in the back side of the tube is drilled, an aluminum patch is bonded in place with EA 9309.1 adhesive.

A typical cross-section of a protruding connector is shown on Figure 18. The suction connection and collar are 2024-T3 aluminum tube. The collar is bonded to the connector with EA 9309.1 adhesive. The flange is eight plies of fiberglass per MIL-C-9084: fabric 181-75 DE. Bonding is accomplished using FM 123-2.

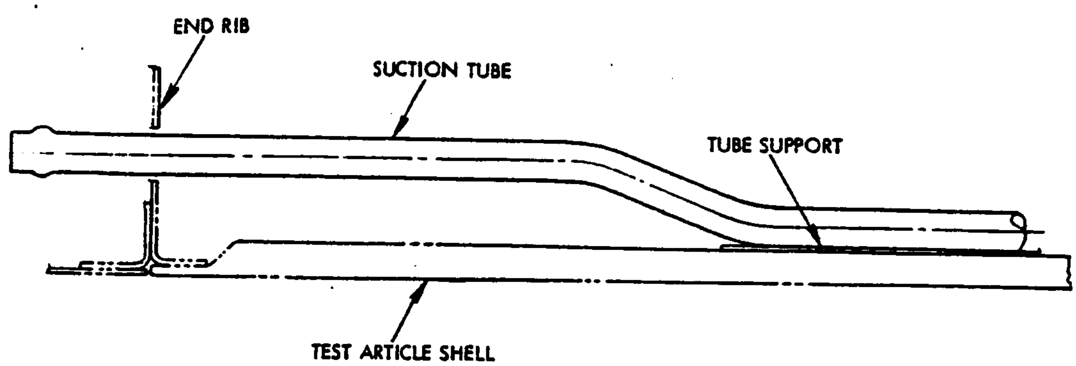

Figure 16. Surface Tube Installation

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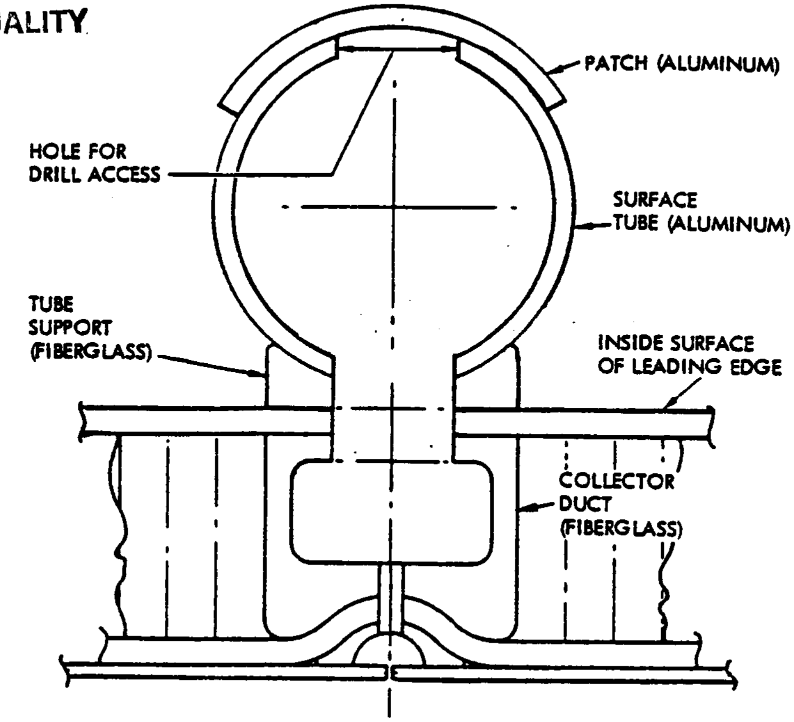

Figure 17. Surface Tube Section

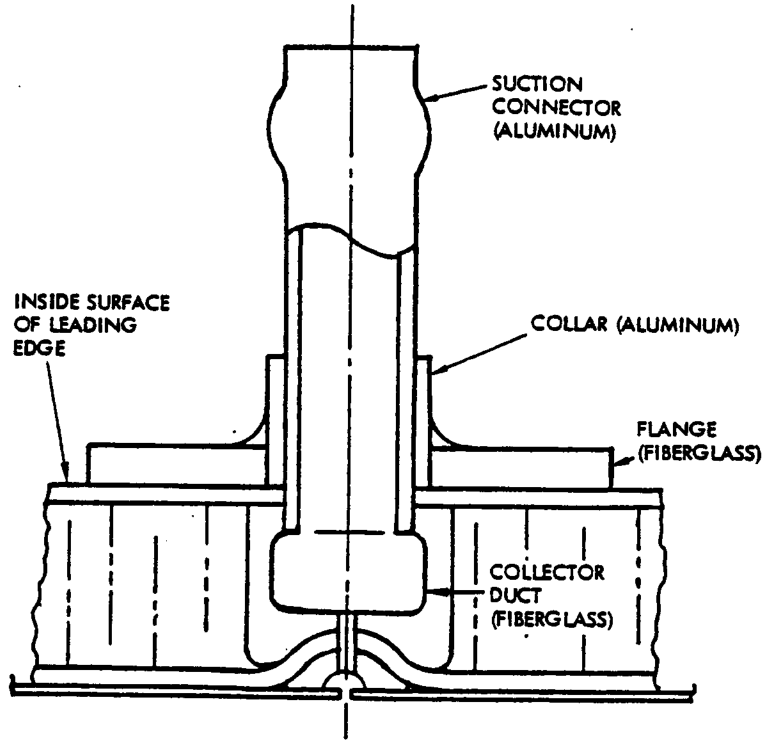

Figure 18. Protruding Connector Section

The two forward-most leading-edge slots are designated for cleaning only, (C1 and C2) having only enough suction applied to prevent back flow when in the LFC suction mode. These slots have restrictors installed in the protruding connectors as shown in Figure 19. The restrictors meter the liquid flow for the cleaning/anti-icing mode of operation. Restrictors are made from 2024-T3 aluminum and are bonded in place with EA 9309.1 adhesive.

Protruding connectors are connected to piccolo suction tubes, which in turn, connect to the LFC aircraft plumbing system. A typical section along the test article, Figure 20, shows the protruding connector to piccolo connection accomplished with flexible air-hose per MIL-H-5593 and MIL-H-8794. Miniature hose clamps are installed on each end to assure the hose will remain connected when purge pressure is applied and to assure a tight suction connection.

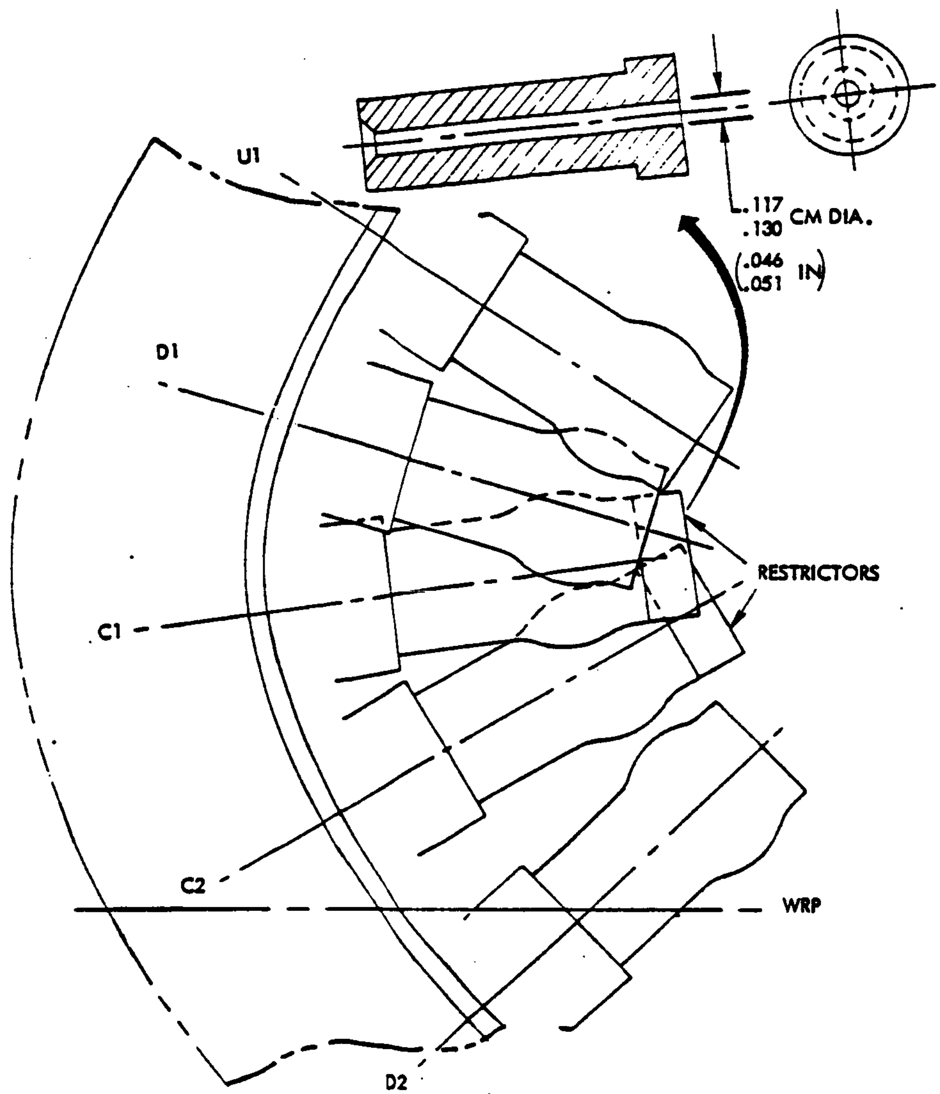

Figure 19. Restrictors

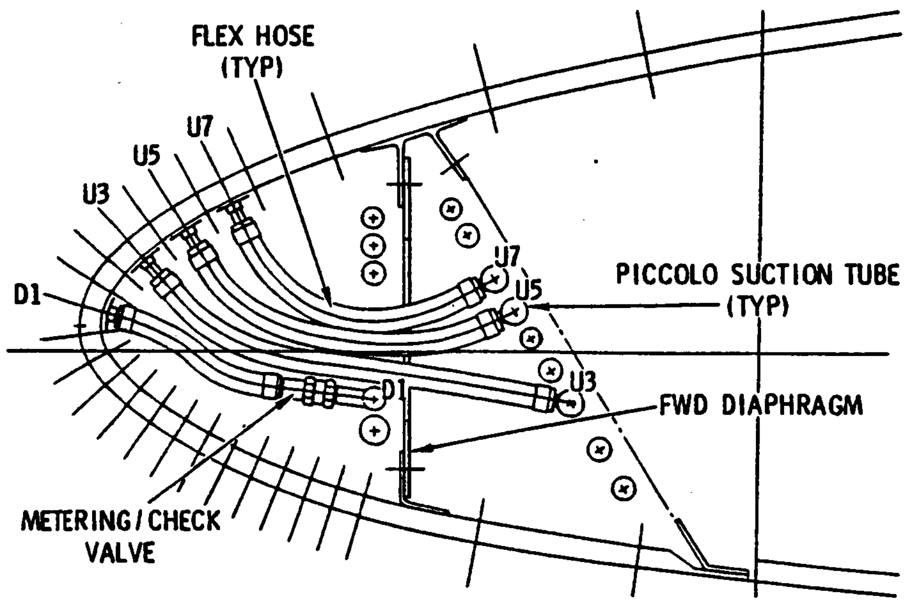

Figure 20. Test Article Section

The connection to piccolo D1, Figure 20, has a metering/check valve installed in the line. This is typical in nine places along each of the six combination suction/cleaning-fluid slots (D1 thru D6). This valve allows use of a single suction/cleaning line to the LFC aircraft plumbing. A metering hole is bored through the check valve poppet to meter the cleaning/anti-icing fluid in that mode of operation. The poppet opens and allows full flow through the valve when the suction mode is selected. The metering/check valve, shown in Figure 21 is made from a NUPRO Company A-4CP2-1 check valve. One end of the valve body is turned down to fit into the piccolo tube connector, and a specially made fitting is installed in the opposite end to fit the flex air-hose. The spring and end seal "O" ring are removed from the valve and a 0.099/0.112 cm (0.039/0.044 in) diameter hole is drilled through the head of the poppet. The valve body and poppet are aluminum; the special end fitting is 2024-T3 aluminum.

Figure 22 shows a composite piccolo tube with a typical direct hose connector and a through-the-metering/check valve hose connector. The connectors and main tube, made from 6061-T6 aluminum, are welded together to form the main body of the assembly. The end plug is made from 6061-T651 aluminum. The end plug and the metering check valve are bonded in place using adhesive EA 9309.1. Piccolo tubes D1 thru D6 have metering/check valves, while C1, C2, U1, U2, U3, U5, and U7 have no metering/check valves.

To install the suction tubes in the limited space of the test article and to better match the tube pressure drop in relation to spanwise pressure gradients, some suction tubes are routed out of the outboard end of the test article and then routed inboard through the test article behind the aft diaphragm, as shown in Figure 23. Figure 24 shows one of the spanwise supports for the return tubes as well as piccolo tube supports. There are five sets of these support blocks in the test article along its span supporting the suction tubes. Support assemblies are made as matched parts from 2024-T351 aluminum.

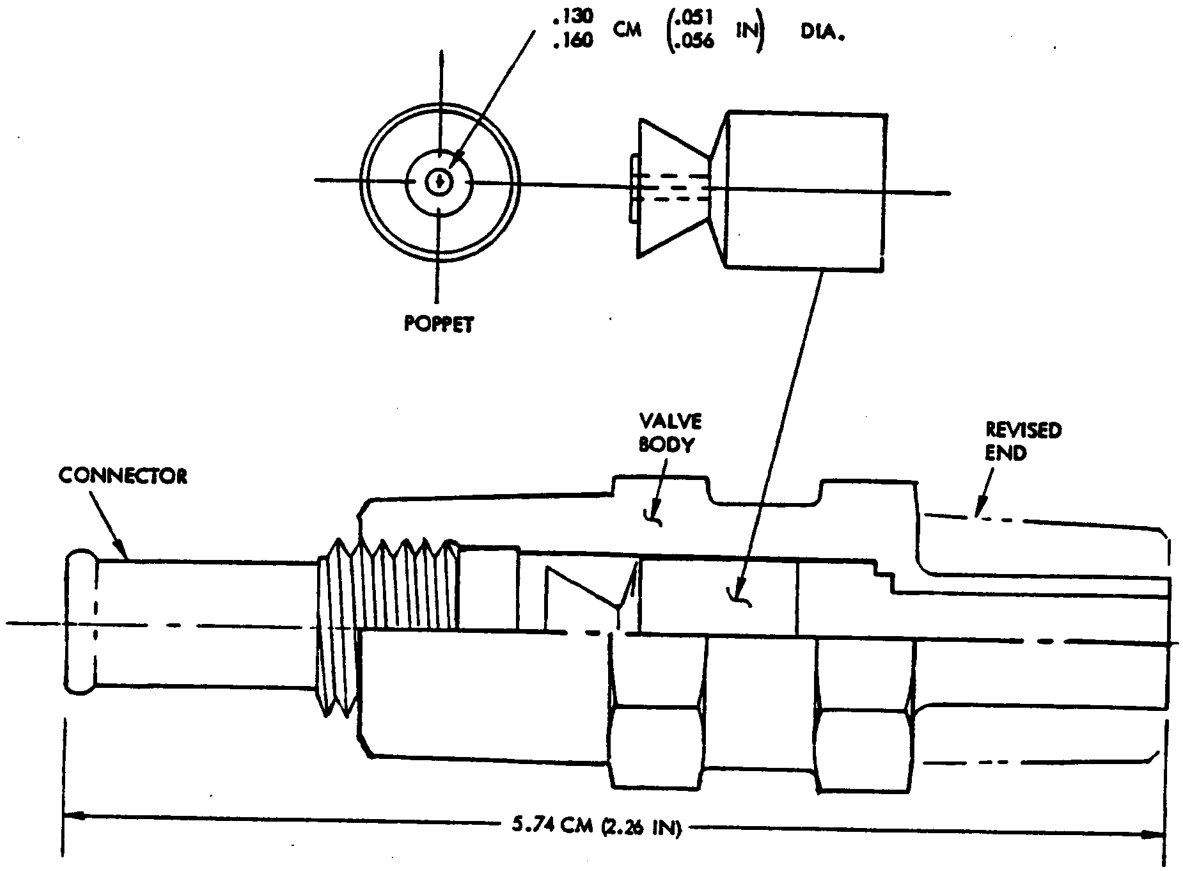

Figure 21. Metering/Check Valve

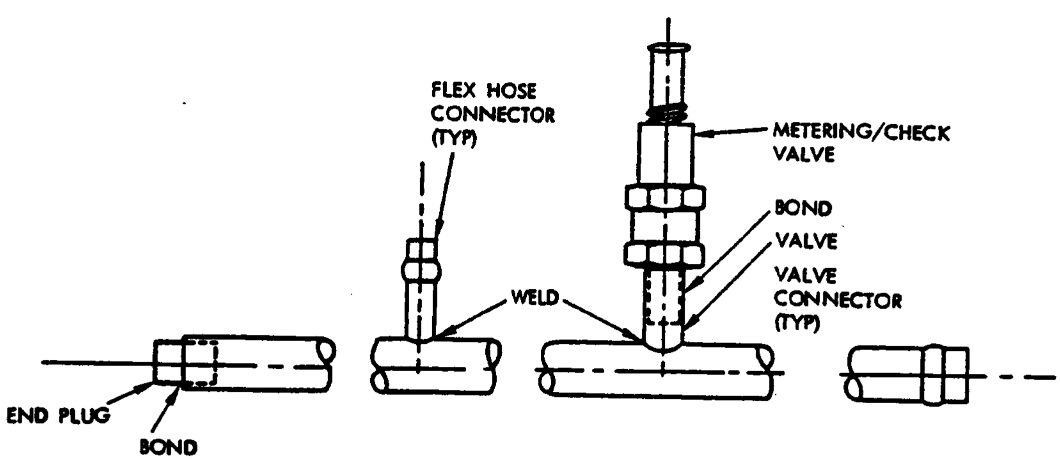

Figure 22. Piccolo Tube Assembly

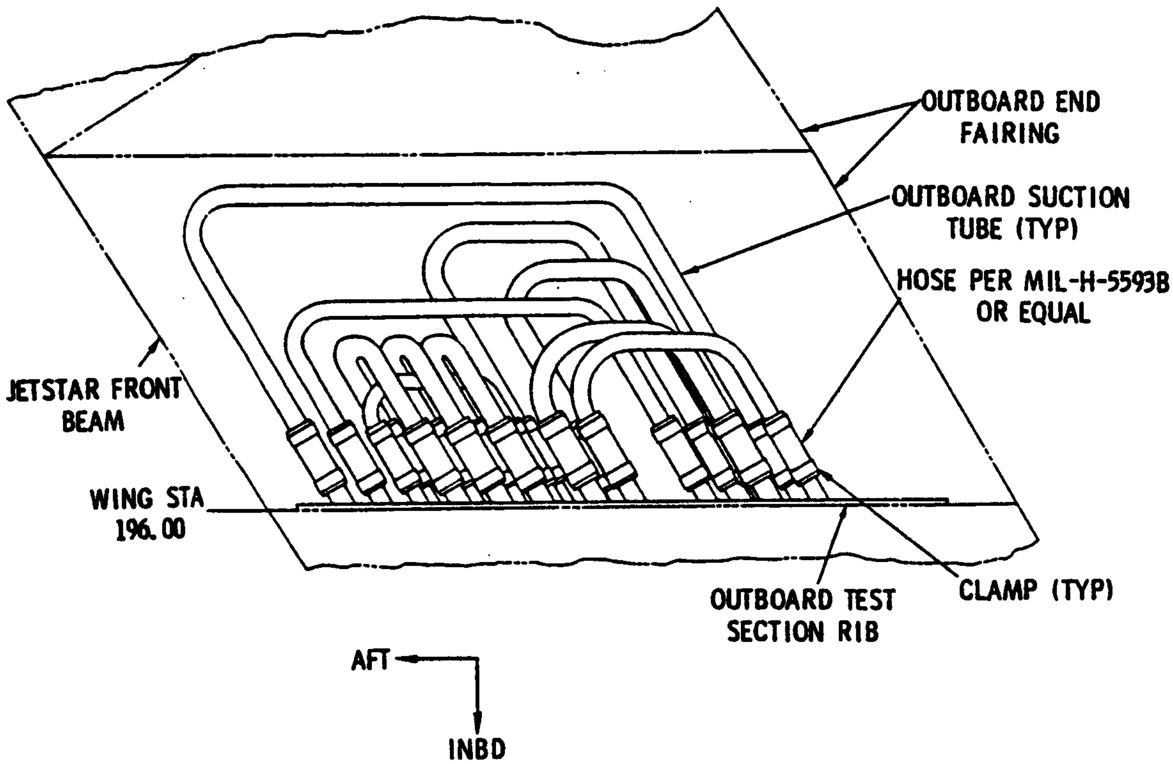

Figure 23. Outboard Tube Connectors

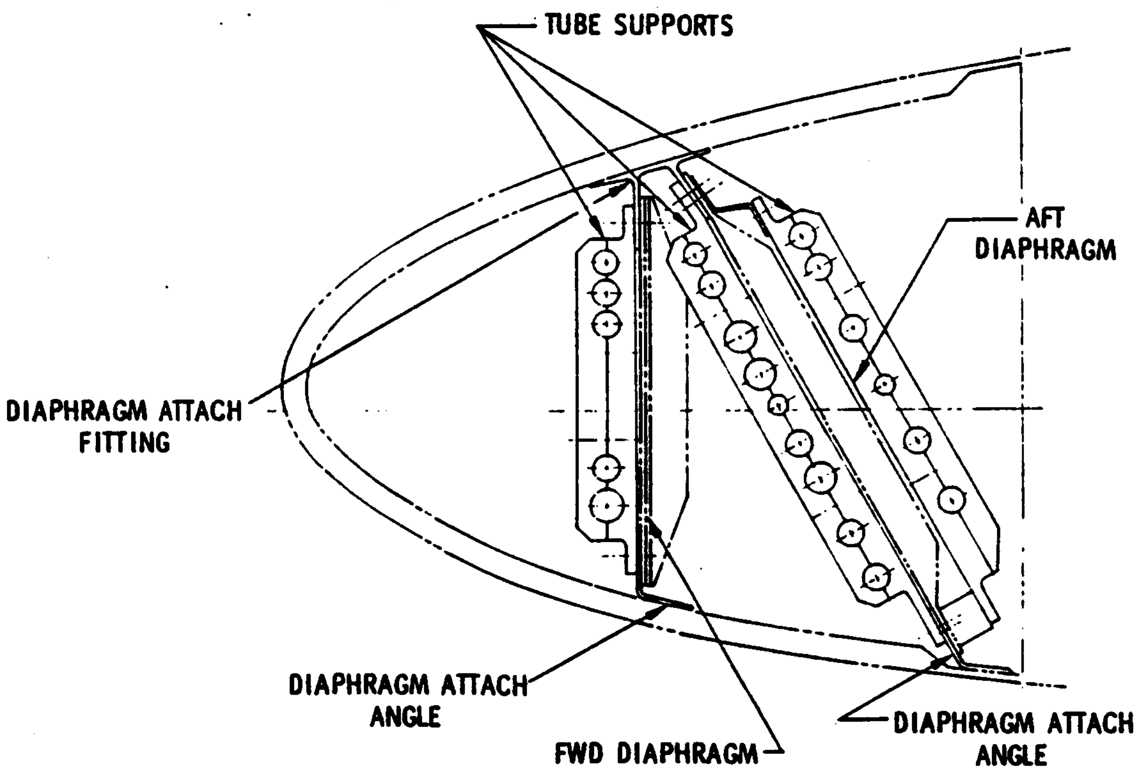

Figure 24. Tube Supports

The internal ducting of slots C1 and C2 is combined into a common piccolo tube and routed directly to the test article inboard end as are U2 through U11 and L1, L2, L3, L7, and L8. Tubes U1, L4, L5, L6, and D1 through D6 are routed to the outboard of the test article, turned and routed back through the aft section of the test article to the inboard end. There are 26 suction tubes at the inboard interface.

4.1.3 Instrumentation

Instrumentation installed in the test article consists of pressure taps and hot-film sensing units to determine the surface Cp distribution, the spanwise slot flow distributions and location of boundary-layer transition. All instrumentation lines are routed to the inboard interface of the leading-edge test article. Instrumentation lines are identified by a alphanumeric number consisting of the designation of the slot with which they are associated, or the slot immediately forward of the location of surface taps. These two digits are followed by an alpha identification of the approximate spanwise location (A through E) identified in Figure 25. In case of collector duct pressure taps, the first two digits are followed by a "C" collector duct - followed by an alpha identification of the approximate spanwise location.

4.1.3.1 Collector Duct Pressure Taps

Eight selected dedicated suction slots (U1, U2, U4, U6, U8, L1, L3, and L5) have static pressure taps installed in both the inboard and outboard ends of the collector ducts. These are in the form of 0.071 cm (0.032 in) OD - 0.051 cm (-0.020 in) ID stainless-steel tubes with the inner end located approximately flush with the inner wall of the collector duct at wing stations 138.00 and 191.00 $\pm$2.54 cm (1 in), which is beyond the blocked end of the slot/slot duct. The inboard spanwise location is coded "A" and the outboard location coded "E" on the tap identification tag; for example, "U1CA."

Figure 26 shows a typical collector duct pressure tap. The instrumentation tube is bonded to the test article inner face sheet and in position using EA 9309.1 adhesive. The tube is CRES-Annealed Hypoflex.

4.1.3.2 Surface Pressure Taps

Surface pressure taps are located in line with and just beyond the ends of selected slots on both upper and lower surfaces at A and E locations shown in Figure 25. These taps are aligned approximately chordwise across the airfoil surface. These taps are located at wing stations 137.60 and 192.46, and the inboard end of the tubes at the end of the test article are appropriately identified by slot number and "A" and "E", such as U4A or U3E. The slots with this instrumentation are as follows:

Upper Surface	Lower Surface
D1, U1, U2, U3, U4	C1, C2, D2, D3, D4
U6, U8, U9, U10	D5, L1, L3, L5, L7
U10.5*, U11	L8

*A pressure tap is installed on both inboard and outboard ends half way between U10 and U11 designated U10.5A and U10.5E.

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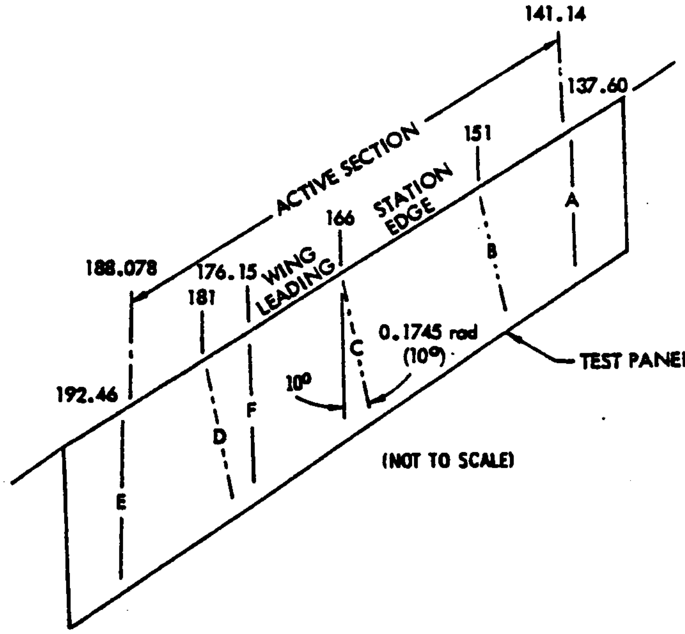

Figure 25. Leading Edge Test Article Instrumentation

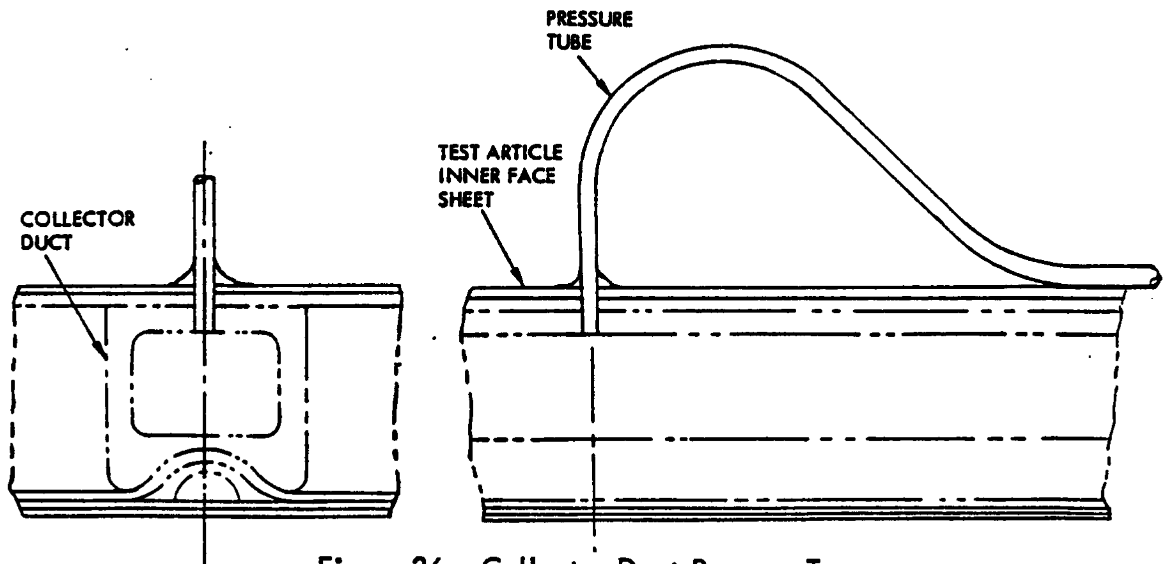

Figure 26. Collector Duct Pressure Tap

The test article shell core is potted with EA 934 adhesive per FED Spec MMM-A-132 for installation of U10.5A and U10.5E. Figure 27 shows a typical off-slot-end surface pressure tap. The tube is bonded to the test article inner face sheet and in position using EA 9309.1 adhesive. The tube is 0.081 cm (0.032 in) OD - 0.051 cm (0.020 in) ID CRES-Annealed Hypoflex.

Static pressure taps are installed approximately midway between selected suction slot pairs on the active suction surface. These taps between the slots are located in three chordwise rows at wing stations 151, 166 and 181 at x/c =

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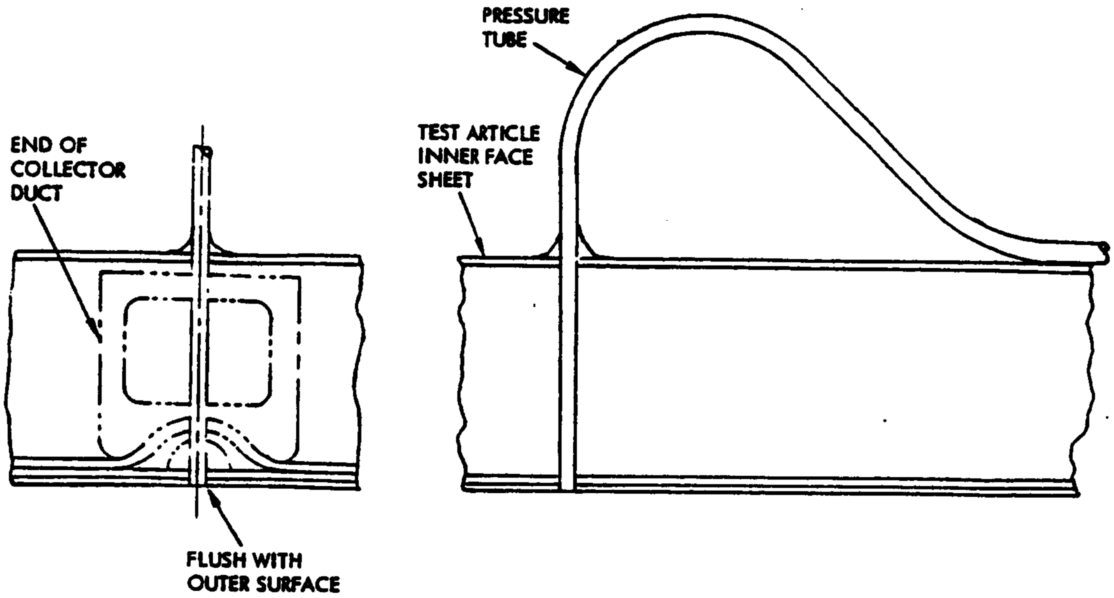

Figure 27. Off Slot End Pressure Tap

0.000 and aligned 0.174 rad (10 deg) inboard relative to the streamwise direction. Pressure taps in these rows are designated B, C, D as shown in Figure 27. Taps are located along these alignments between the following pairs of slots.

Upper Surface	Lower Surface
U3 - U4	D3 - D4
U4 - U5	D4 - D5
U5 - U6	D6 - L1
U7 - U8	L2 - L3
U8 - U9	L4 - L5
U9 - U10	L6 - L7
U10 - U11	L7 - L8

All 42 taps located within the active suction area are constructed by penetrating the test article titanium surface with 0.042 cm (0.018 in) OD - 0.025 cm (0.010 in) ID CRES - Annealed Hypoflex tube. These taps are installed to maintain critical surface smoothness. An 0.071 cm (0.032 in) OD - 0.051 cm (0.020 in) ID CRES-Annealed Hypoflex tube is silver-brazed over the end of the smaller pressure tap tube as near the wing surface as practical and is then routed to the inboard end of the test article. The small ID at the surface is required to avoid delaminarizing the boundary layer. The transition to a larger ID as soon as practical will facilitate purging of these lines and will reduce lag time in the data acquisition system. These taps at spanwise locations are identified by the designation of the slot immediately forward of the tap location followed by the appropriate alpha designation for the spanwise location: U5C or L4B, for example.

Figure 28 shows a typical between-duct pressure tap installation. Bonding of the between duct pressure tubes is accomplished with EA 9309.1. There are 10 pressure taps in the over-wing fairing for the LFC JetStar, designated T1 through T10. They are on a line slanted 0.209 rad (12 deg) inboard from a point on the JetStar 12 percent wing chord at wing station 160.55. The first tap location is 6.35 cm (2.50 in) aft of the 12 percent line and the last 68.58 cm (27.00 in) aft along the tap line.

4.1.3.3 Hot-Film Sensors

Six hot-film sensors are installed in the upper surface of the test article. Six additional sensors are located in the over-wing fairings.

Sensors in the test article are located along wing station 176.15 and are spaced chordwise with two sensors located between slots U9 and U10, one sensor aft of slot U11, and one sensor between each of the following pairs of slots U5 and U6, U8 and U9, and U10 and U11. The six sensors in the over-wing fairing locations are on 13.62 cm (5.40 in) centers along the wing station 176.15 line with the first sensor located 9.53 cm (3.74 in) aft of the 12 percent chord. The sensors are TS1 Model 1268 sensors installed flush with the wing surface. The test article shell is potted for hot-film sensor installation where required with EA 934 adhesive.

The hot-film sensors are wired to the inboard end of the test article using coaxial cable, which is bonded to the test article shell inner face sheet with EAC 9309.1 adhesive. The inboard end of the coaxial cable is identified for its particular sensor. The sensors are identified by an "HF" code followed by a sensor number from 1 to 12, beginning with the most forward sensor: HF1 or HF6, for example. Figure 29 shows the approximate locations of the forward most sensor installation.

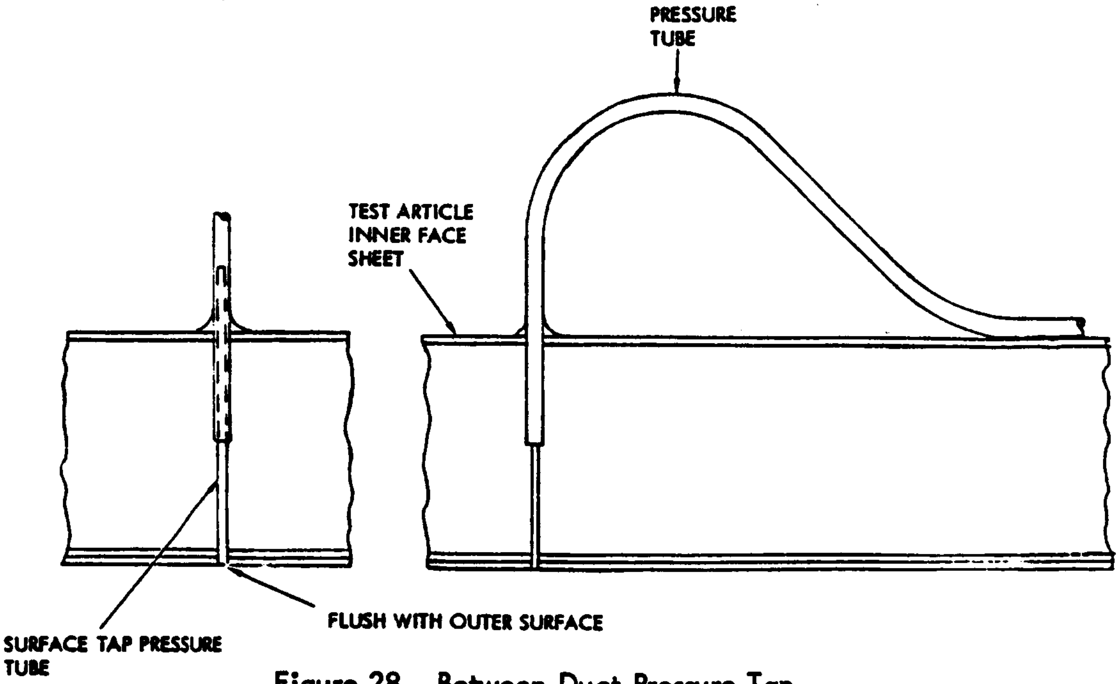

Figure 28. Between Duct Pressure Tap

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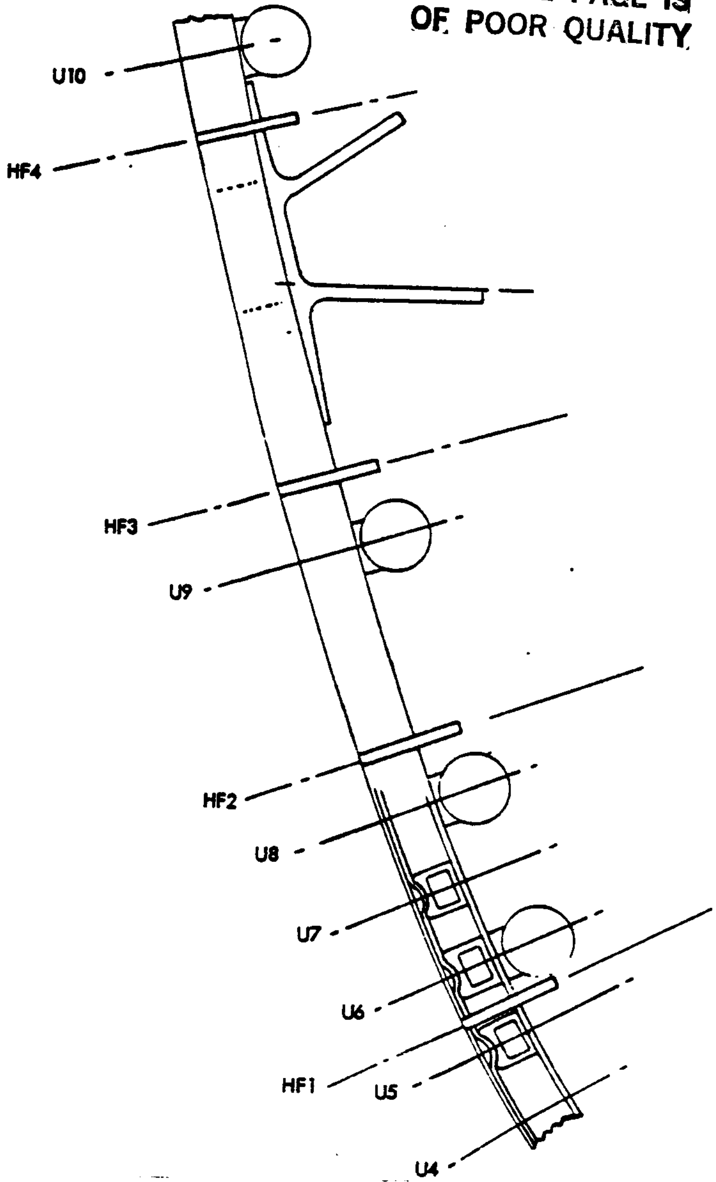

Figure 29. Hot Film Sensor Installation

4.1.4 Installation of Test Article

Since the LFC-LEFT test article is thicker than the JetStar wing at the 12 percent chord attaching line, beam cap extensions are installed on the top and bottom of the wing front beam. Screws through the aft edge of the test article and beam cap extensions provide the primary section support. Floating nutplates are installed on the beam cap extensions, and loose holes are provided in the extension mounting flanges. This allows for differential expansion between the JetStar wing and the test article while providing adequate retention of the section. Figure 30 shows the Lockheed test article installation on the left wing of the JetStar.

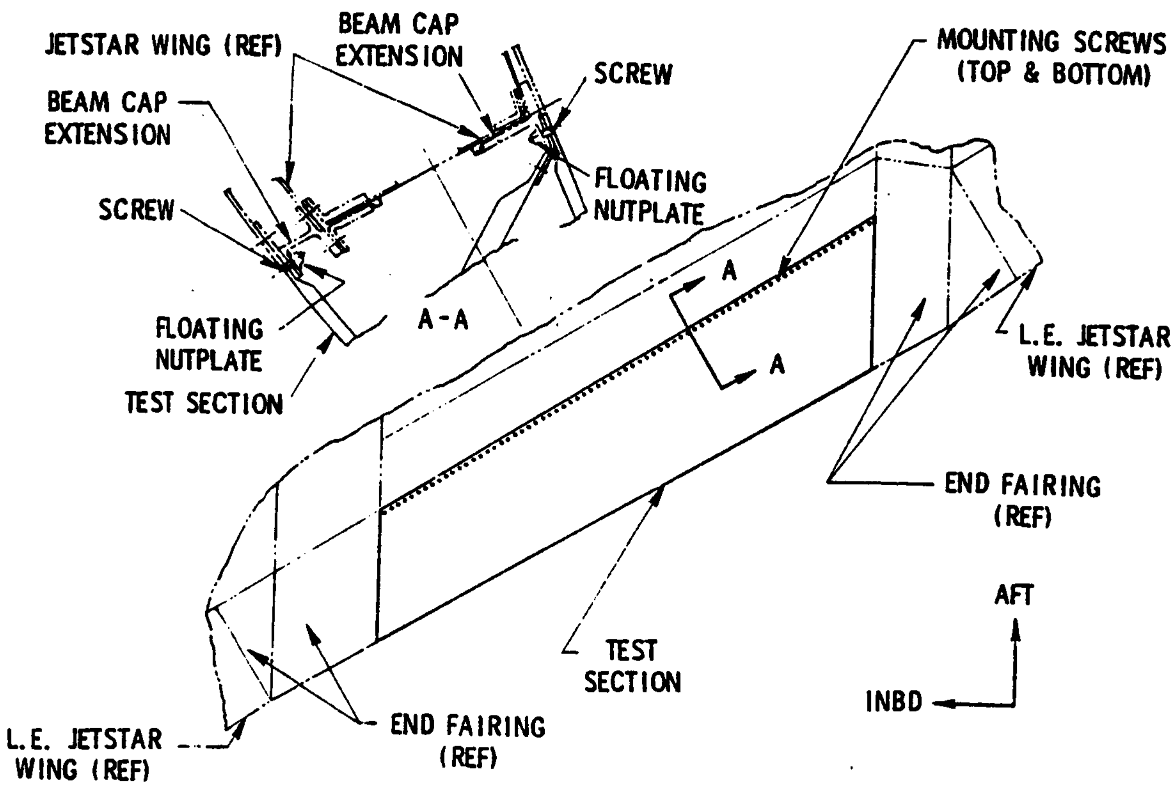

Figure 30. Lockheed Test Article Installation

The McDonnell Douglas LFC-LEFT test article is installed on the right JetStar wing and unlike the Lockheed test article it is supported by ribs at three positions along its length. There are double ribs at JetStar wing leading-edge stations 168.77 and 213.33 with a single rib at leading-edge station 194.48. The double ribs are used to support actuators for the LFC leading-edge shield, a part of the McDonnell Douglas design. Machined fittings are installed on the wing beam with permanent fasteners and the support ribs are fastened to the machined fitting by removable fasteners. Installation of the McDonnell Douglas test article is shown in Figure 31.

The Lockheed LFC-LEFT test article is properly aligned and installed on the JetStar wing with the use of two alignment templates as shown in Figure 32. The templates rest on the wing rear-beam lower cap on the aft end and index to the front-beam lower cap through the lower-beam cap extension and lower LFC fairing. This locates the leading-edge test article to the lower caps of the wing front and aft beams.

To properly align the McDonnell Douglas test article, which provides suction on the upper surface only, separate alignment pieces are added to the front of the alignment templates with close tolerance pins. This allows for a template bearing point on the upper leading-edge surface. Figure 33 shows one of the templates with the added forward piece.

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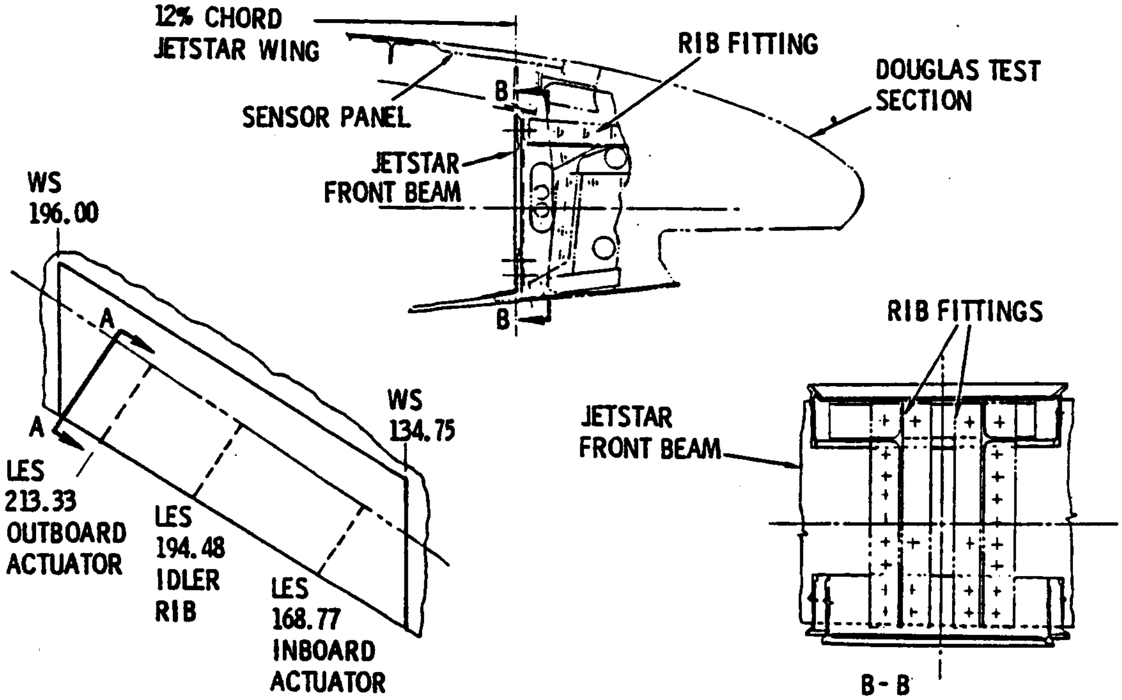

Figure 31. McDonnell Douglas Test Article Installation

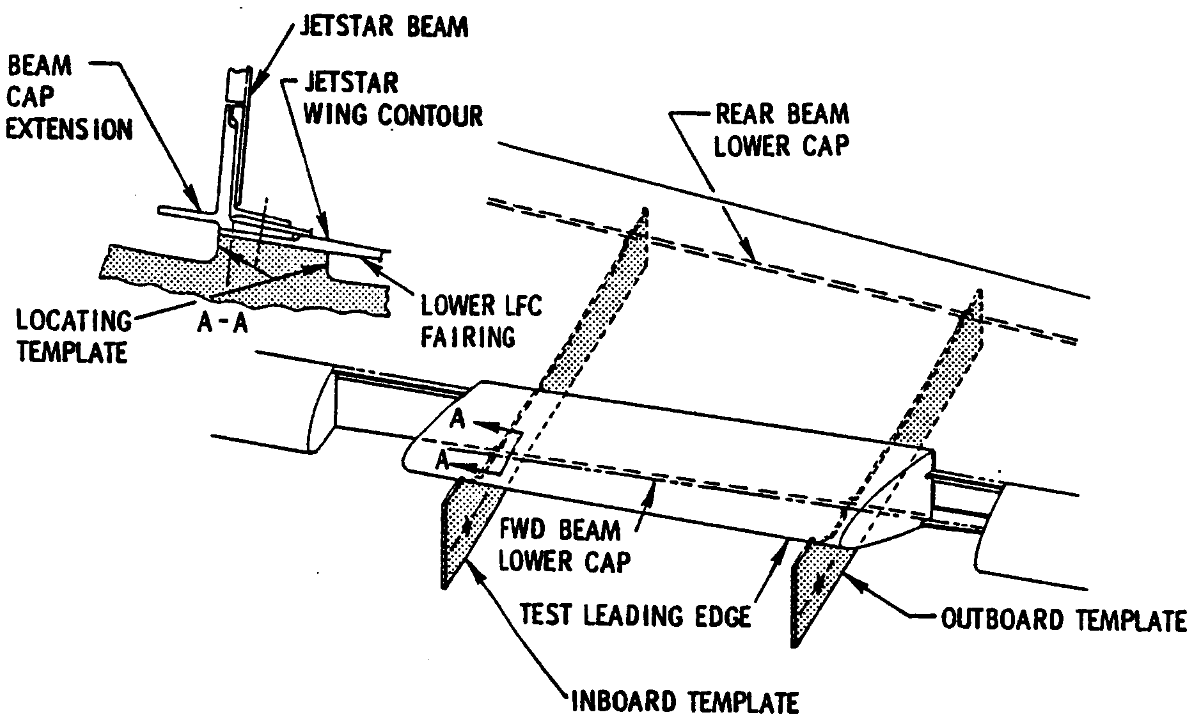

Figure 32. Lockheed Test Article Alignment

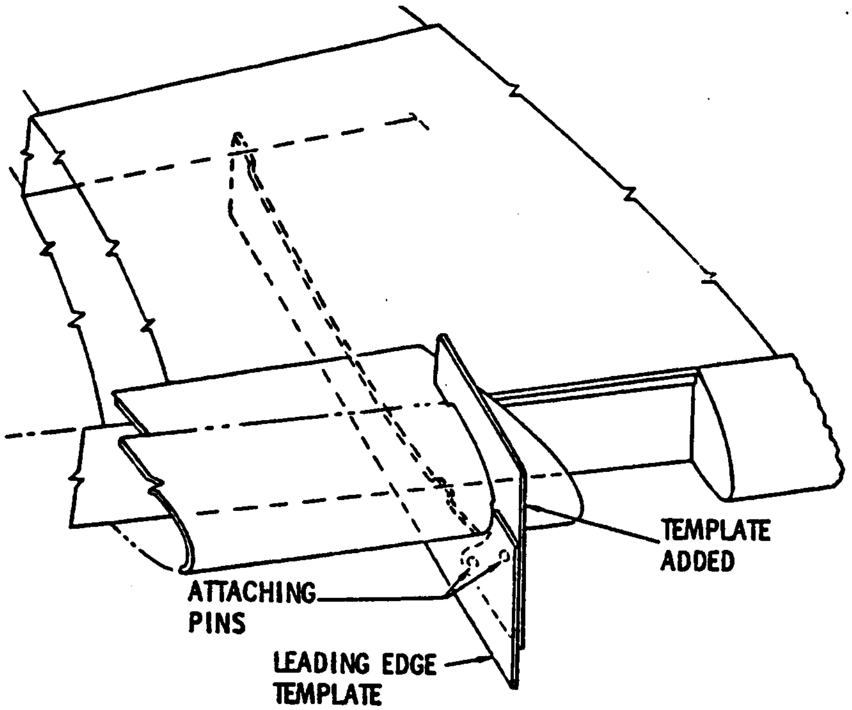

Figure 33. McDonnell Douglas Test Article Installation

4.1.5 LFC Fairings

Figure 34 shows the individual parts of the fairings required for the LFC-LEFT test articles. Except for some of the sub-structure, these fairings are mirror images from the left to the right JetStar wing. The sensor panel for the Lockheed side is shown. On the right-hand wing, this panel is furnished as an integral part of the McDonnell Douglas test article. Except for the inboard triangular piece of skin on the inboard leading-edge fairing which is aluminum, the fairing skins are fiberglass with aluminum formers. The over-wing portion of the fairing is the largest expanse of fairing and is made in four panels.

A smooth surface is maintained at all fairing-to-test article joints. Figure 35 shows a typical cross-section of the transition of the Lockheed leading-edge test article to upper and lower fairings at JetStar wing 12 percent chord. All joints are smooth, and gaps are filled with aerodynamic putty.

Installation of the McDonnell Douglas test article and fairings are similar to the Lockheed installation except that the McDonnell Douglas sensor panel is made an integral part of the test article and is joined to the test article forward of the JetStar 12 percent wing chord line. The aft end of the sensor panel is along the same line as the Lockheed sensor panel and the joint to the over-wing fairing is similar to the Lockheed side. Figure 36 shows a typical section across the McDonnell Douglas fairing/test article joints.

Back-to-back caps are installed at each test article end rib to provide for leading-edge end fairing attachment. A section through the outboard end of the test article to leading edge end fairing is shown in Figure 37.

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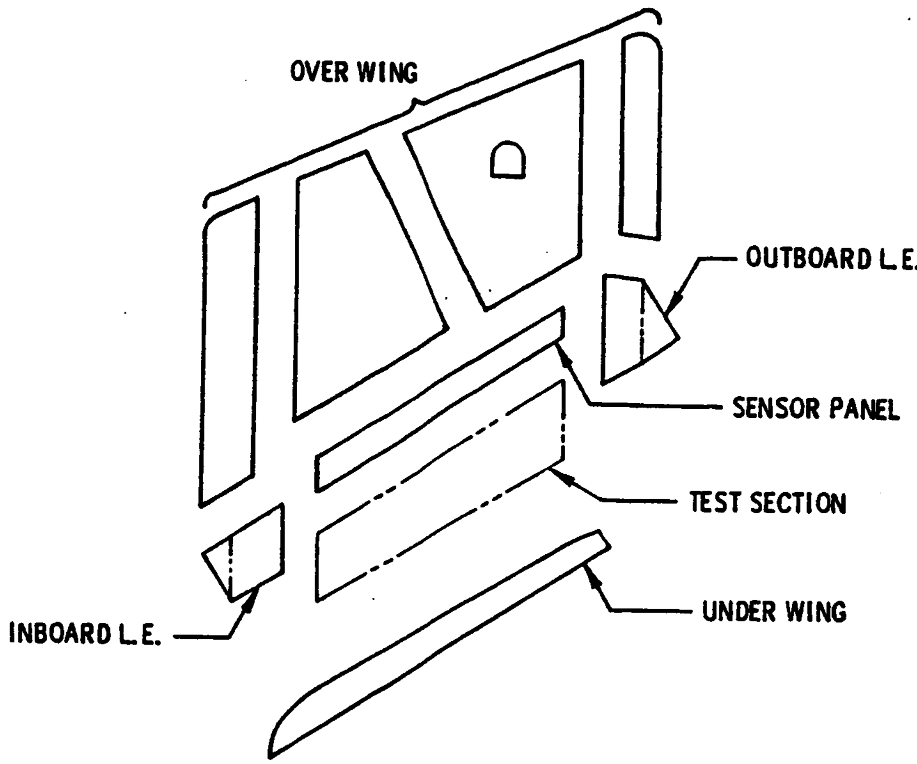

Figure 34. Test Article Fairings

4.1.5.1 Over-Wing Fairing

Formers contoured to the JetStar wing surface on the lower side and to the LFC test contour on the upper side are used to install the over-wing fairings. The formers and fairing edges are attached to the JetStar upper wing panels with screws. The reinforced fiberglass fairings are attached with screws to the upper flanges of the aluminum formers. Shims are provided between the formers and the fairings to allow for adjustment to meet contour requirements. Figure 38 shows the over-wing fairing arrangement. There are two streamwise formers at each end of the test article and eight intermediate formers installed on the wing normal to the JetStar wing center beam. All formers are clipped to the front wing beam upper cap extension. Figure 39 shows a typical former end tie for both left-hand and right-hand fairing installations. The fairing formers are machined from 7075-T7351 aluminum plate, splice plates are 2024 or 7075 clad, and clips are made from 7075 aluminum extrusions.

Local reinforcing intercostals are installed between the integral stiffeners of the JetStar upper wing planks at the former tiedown screw locations. Dome nuts are installed for the tiedown screws to provide sealing for the fuel carried in the wing. A typical installation is shown in Figure 40. The intercostals and end clips are made from 2024-T4 aluminum extrusions.

The over-wing fairing is a twelve ply fiberglass sheet with integral polyurethane stiffeners running spanwise on the fairing. The stiffeners are discontinuous at the over-wing formers. Figure 41 shows the general arrangement of the fairing.

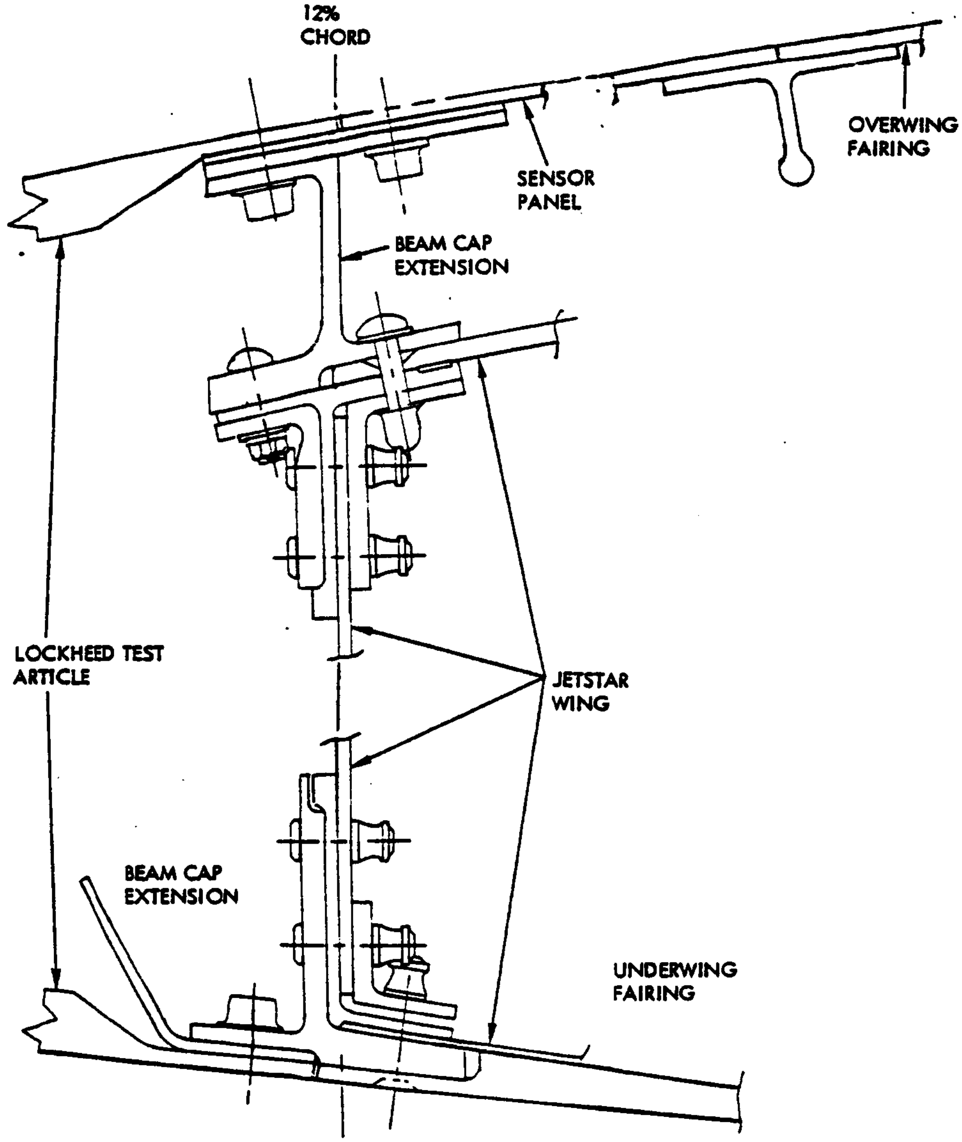

Figure 35. Lockheed Fairing/Test Article Joint

To complete the over-wing fairings, it is necessary to install transition fairings to bring the test section contour back to the JetStar basic wing contour. These transition fairings on each end of the over-wing fairings are of the same basic fiberglass construction as the main fairings. The fairing outer edge fastens directly to the JetStar wing plank. Figure 42 shows a typical cross-section of an outboard transition fairing.

To minimize fairing deflection under air loads, the fairing is vented to upper surface pressure. Figure 43 shows the fairing vent holes.

Adequate cross-ventilation and provisions for water drainage are provided in the fairing substructure. The fairing formers are sealed at the end of the sensor panel at wing station 134.750 and wing station 196.000, and the sub-

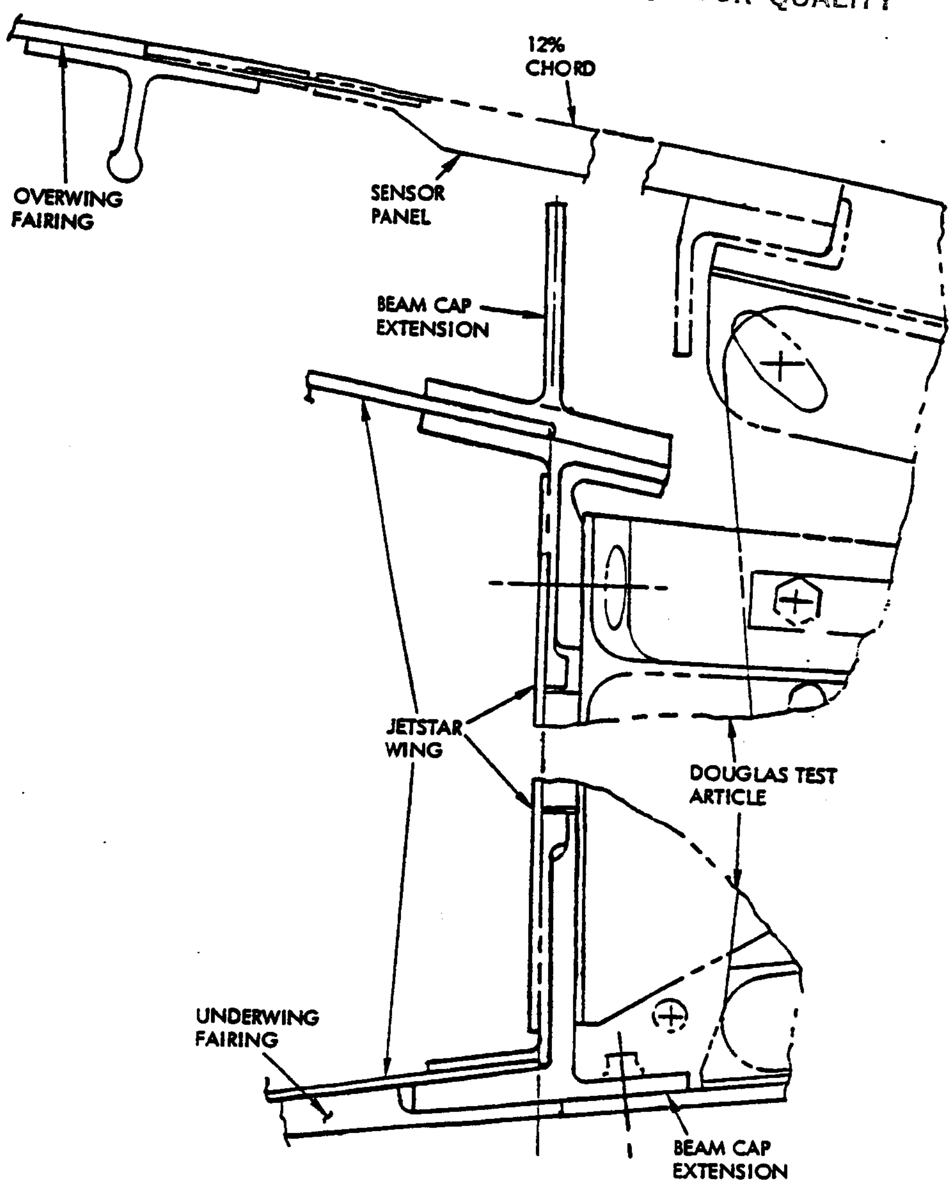

Figure 36. McDonnell Douglas Fairing/Test Article Joint

structure is sealed to the top of the JetStar wing with aluminum angles. The splice tee at the aft end of the sensor panel is used for attachment of the seal angles. Gaps, slots, and holes are filled and overcoated and filleted as required. Figure 44 shows a typical section of sealing in this area.

Since the over-wing fairings cover one of the JetStar wing fuel tank filler openings, access to this opening is provided by a hinged door with a quick-release latch installed in the surface of the fairing. The fuel filler access door is shown in Figure 45. A drain tube is provided for the cavity formed by the pan to carry any spilled fuel overboard.

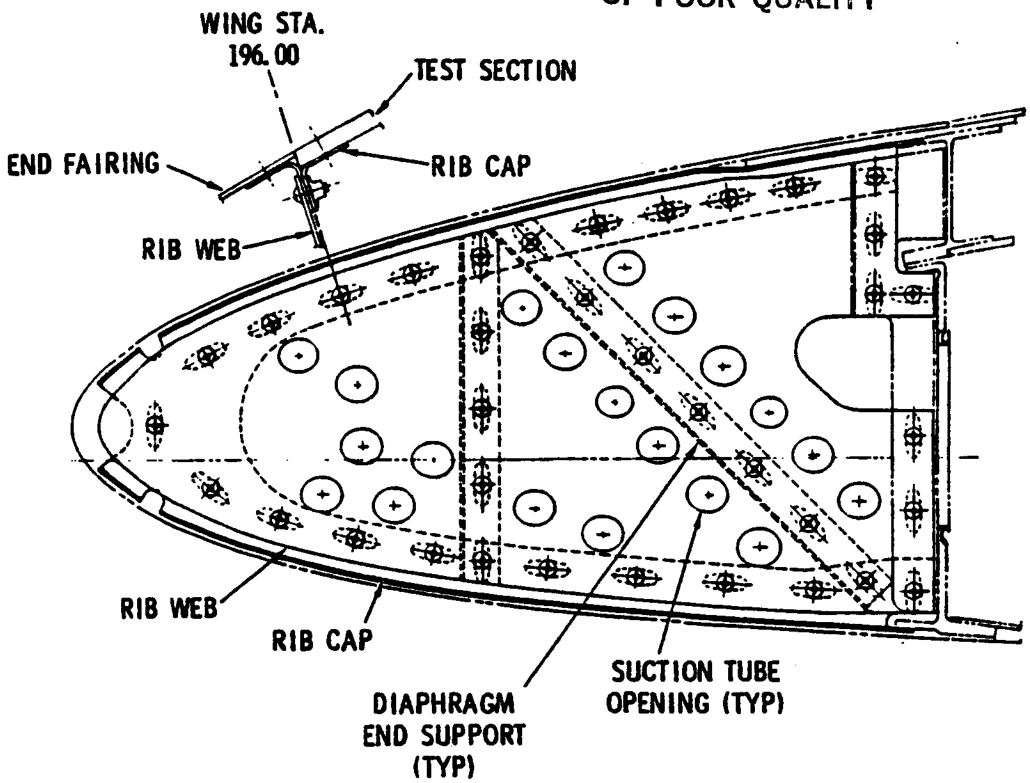

Figure 37. Leading Edge End Fairing Attachment

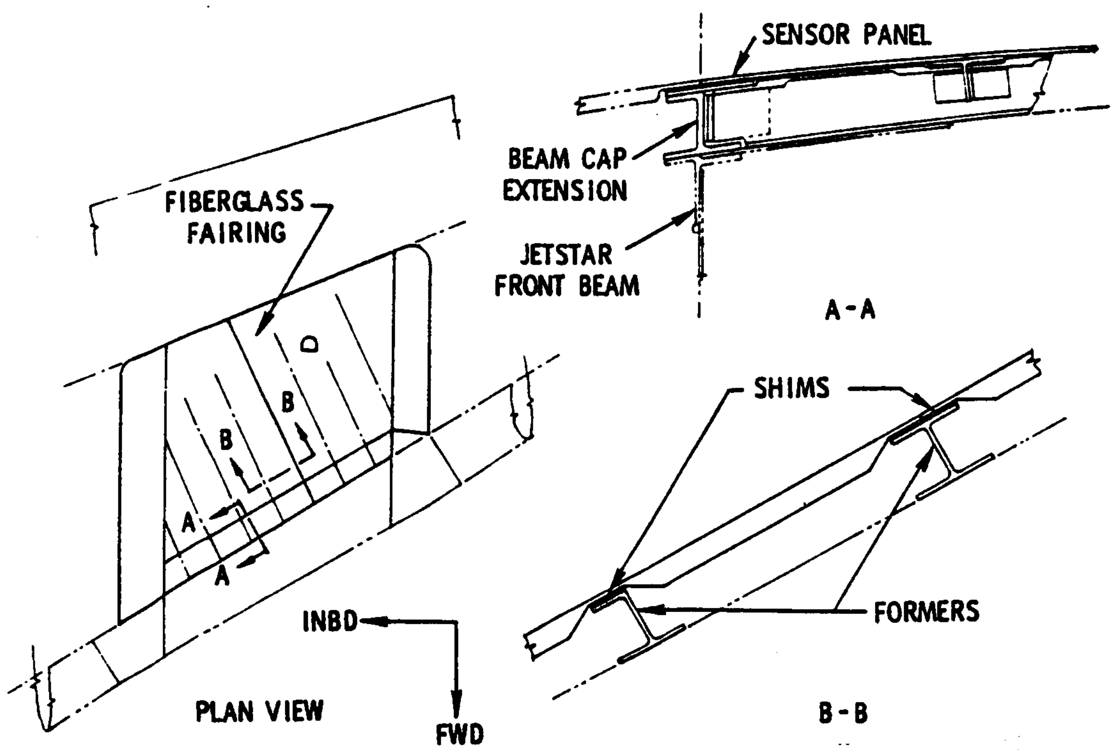

Figure 38. Over-Wing Fairing Arrangement

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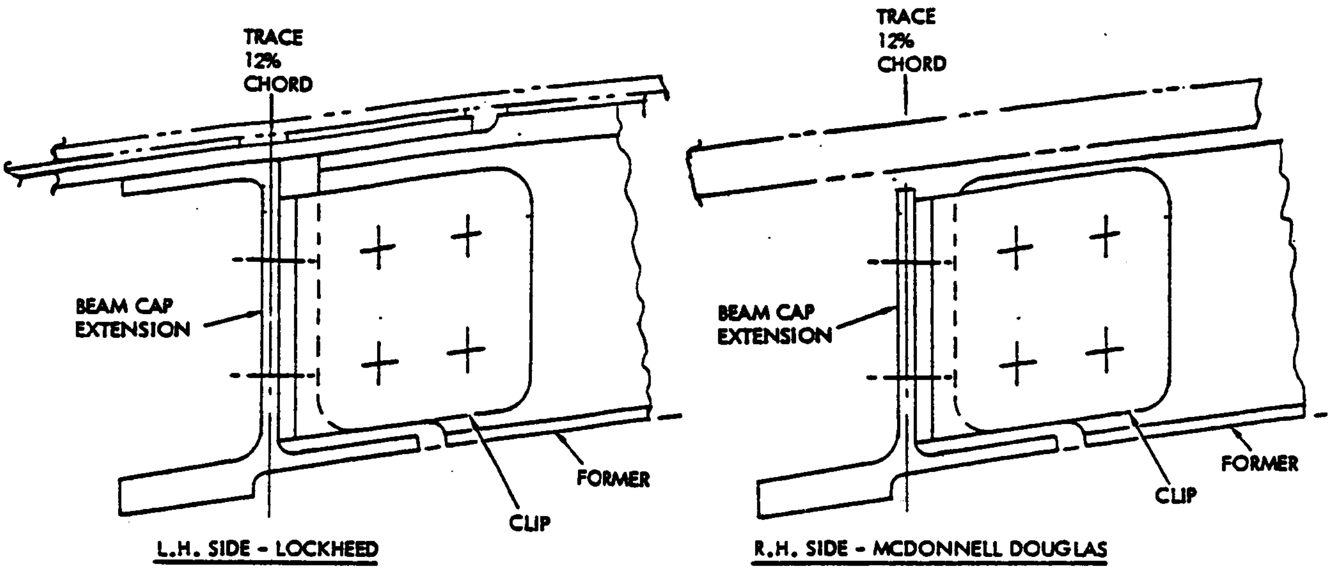

Figure 39. Former End Tie

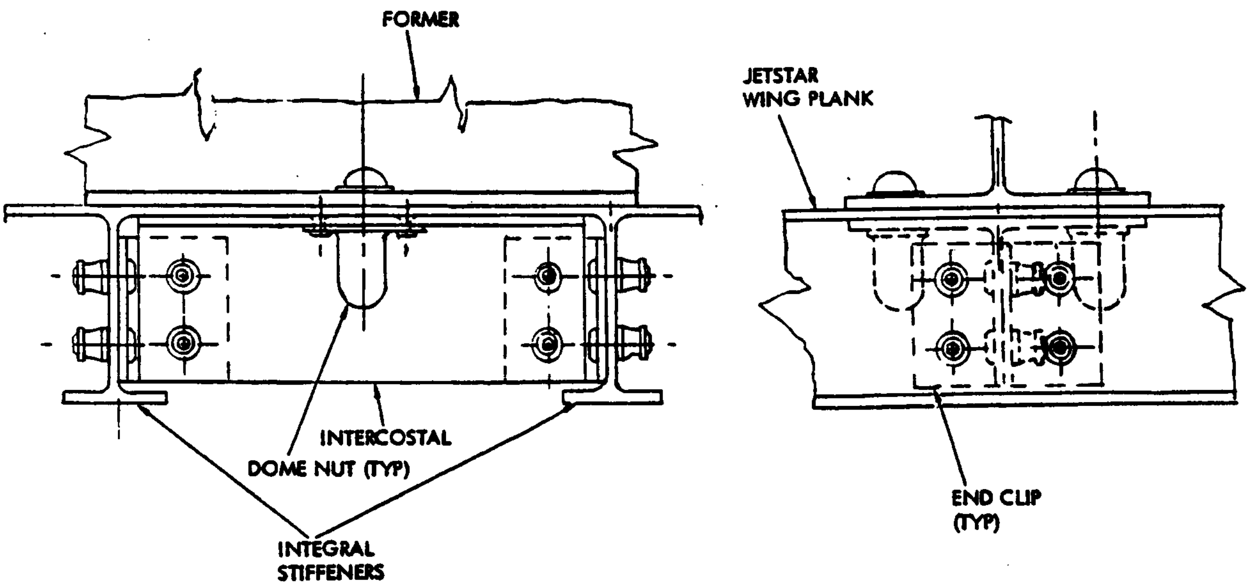

Figure 40. Former/Wing Fasteners

4.1.5.2 Over-Wing Fairing Alignment

To properly align and contour the over-wing fairing, the leading-edge test article must be installed first. With the leading-edge test article in place, three templates are used to control the over-wing fairing installation: one at each streamwise end former and one at the fairing split line former at approximately the center of the fairing. The gap between the template and the former is measured for proper shimming at each of the control positions. All other formers are liquid-shimmed between the fairing and former. Fairing contour between the control positions is checked with a straight edge. Figure 46 shows the end templates in position over the test area.

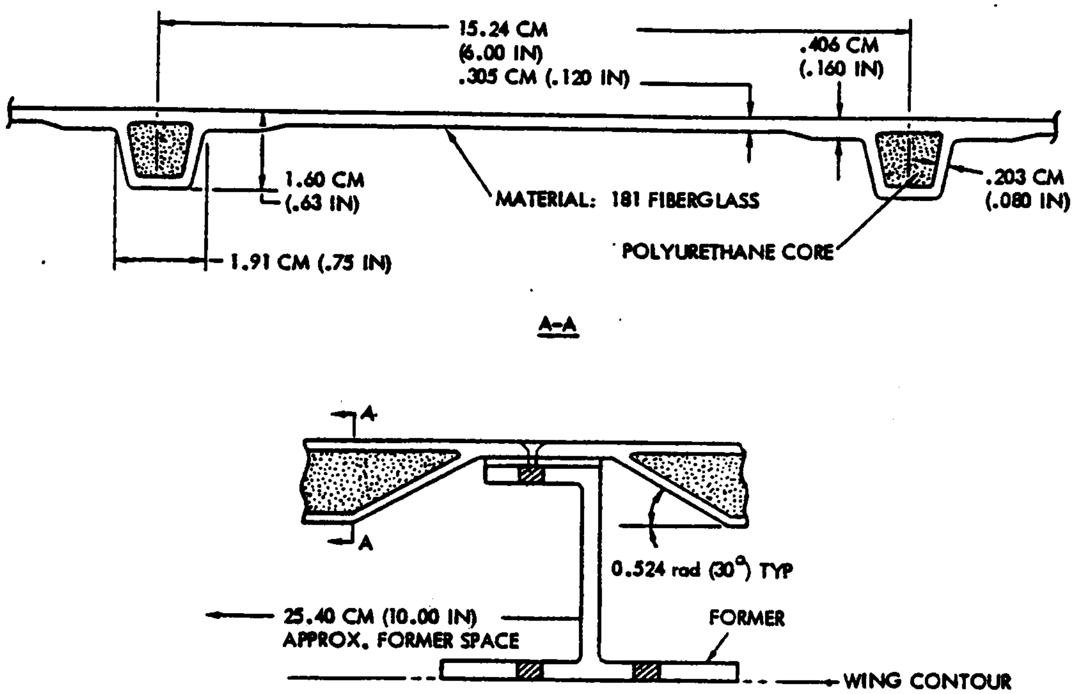

Figure 41. Over-Wing Fairing

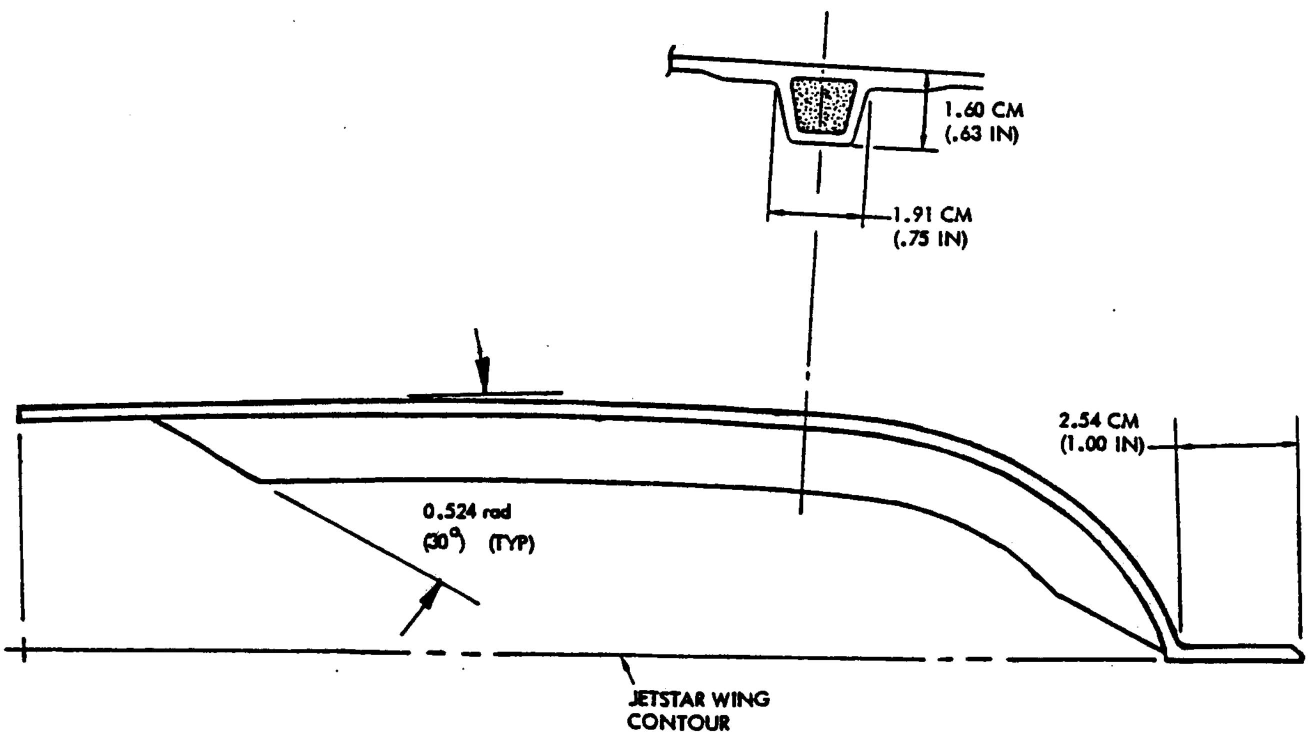

Figure 42. Transition Over-Wing Fairing

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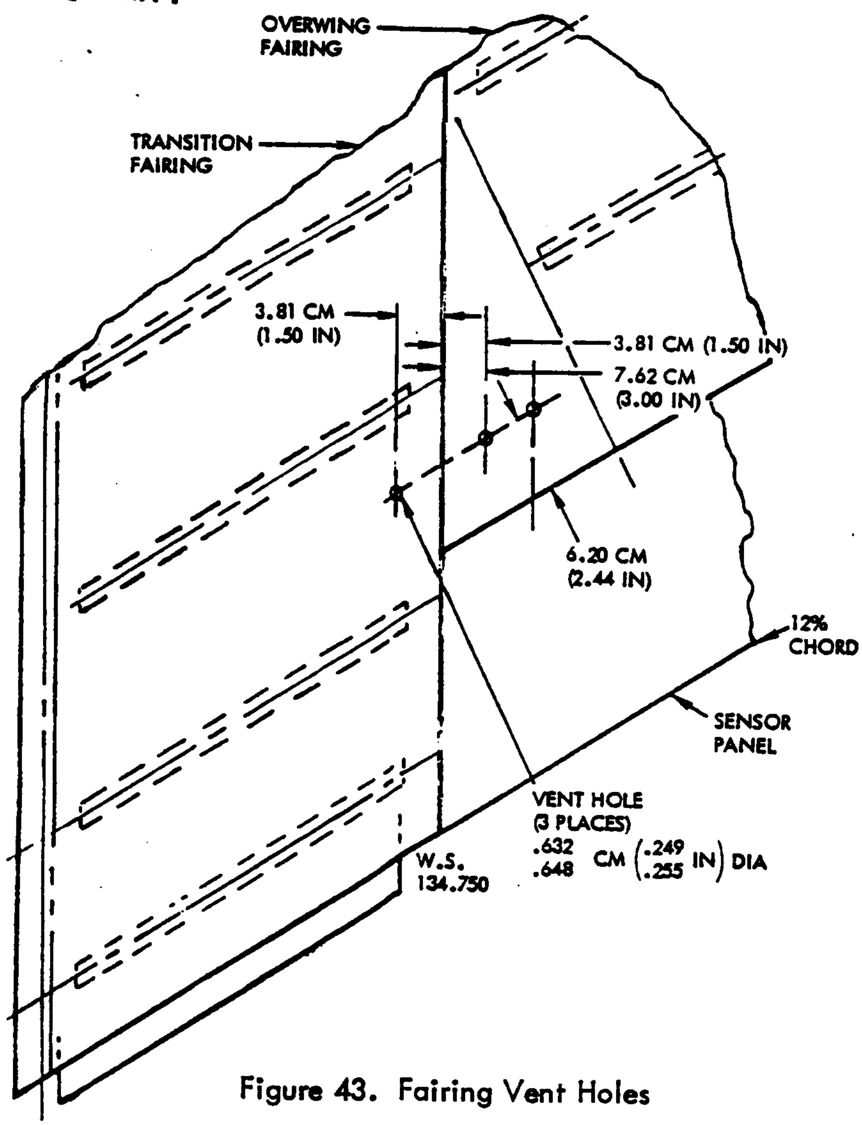

Figure 43. Fairing Vent Holes

4.1.5.3 Under-Wing Fairing

The under-wing fairing is a fiberglass, one-piece, contoured part installed on the bottom of the JetStar wing. Figure 47 shows the under-wing fairing installation.

4.1.5.4 Leading-Edge End Fairing

Removable fairings are installed on each end of the test article to make the transition from the test article contour to the JetStar wing contour.

The fairings are mirror images from left- to right-hand side, except for a revision to the outboard right-hand fairing to provide adequate space for the McDonnell Douglas leading-edge shield drive motor. Figure 48 shows the test article end fairings.

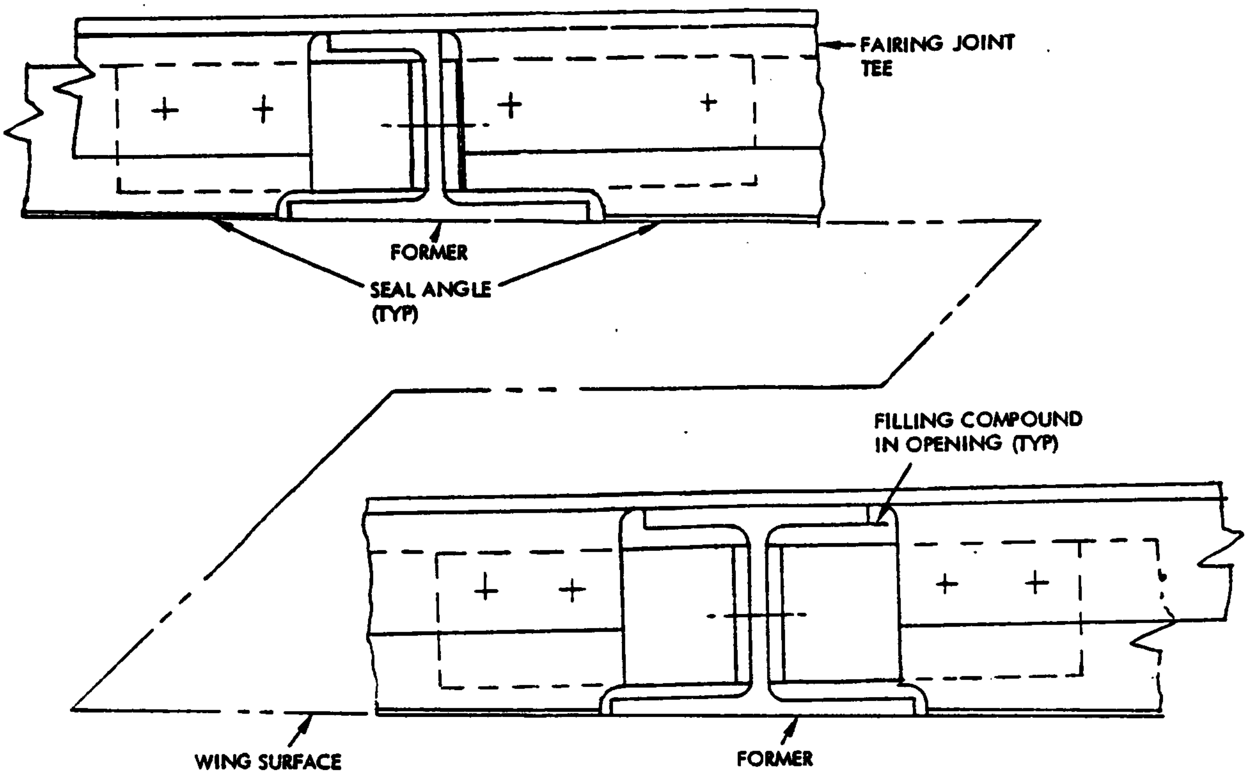

Figure 44. Fairing to Wing Sealing

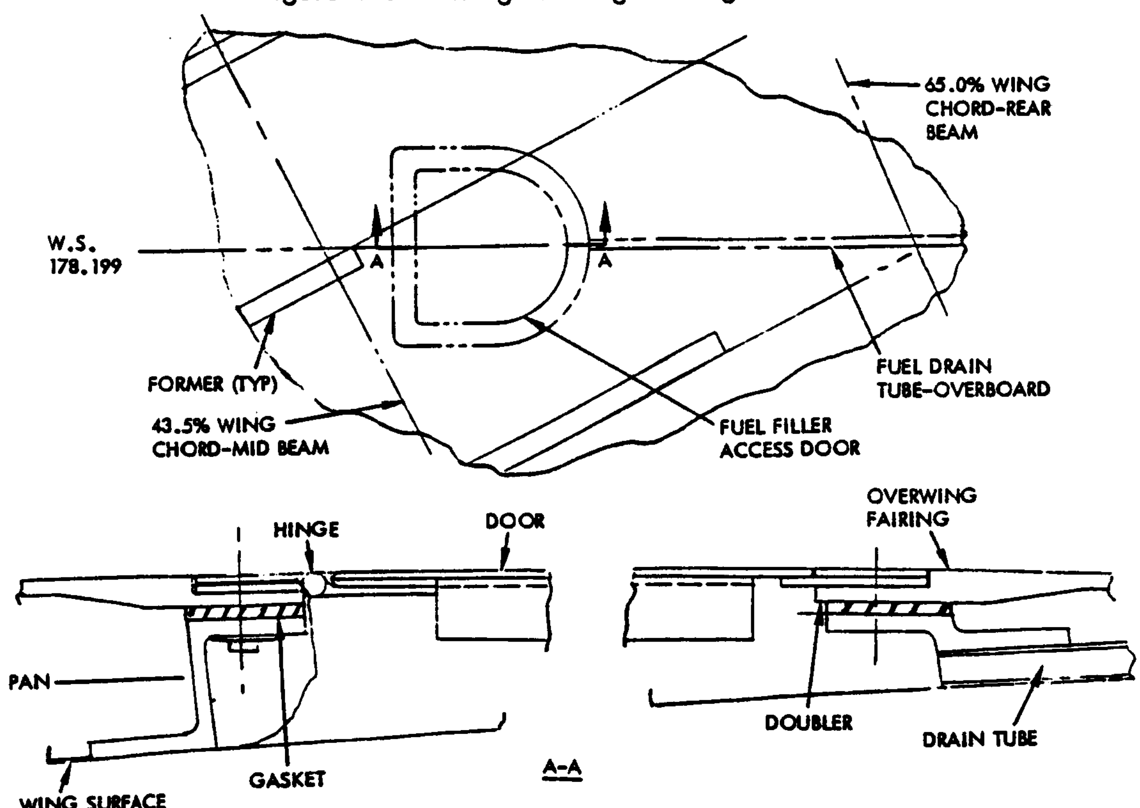

Figure 45. Fuel Filler Access Door

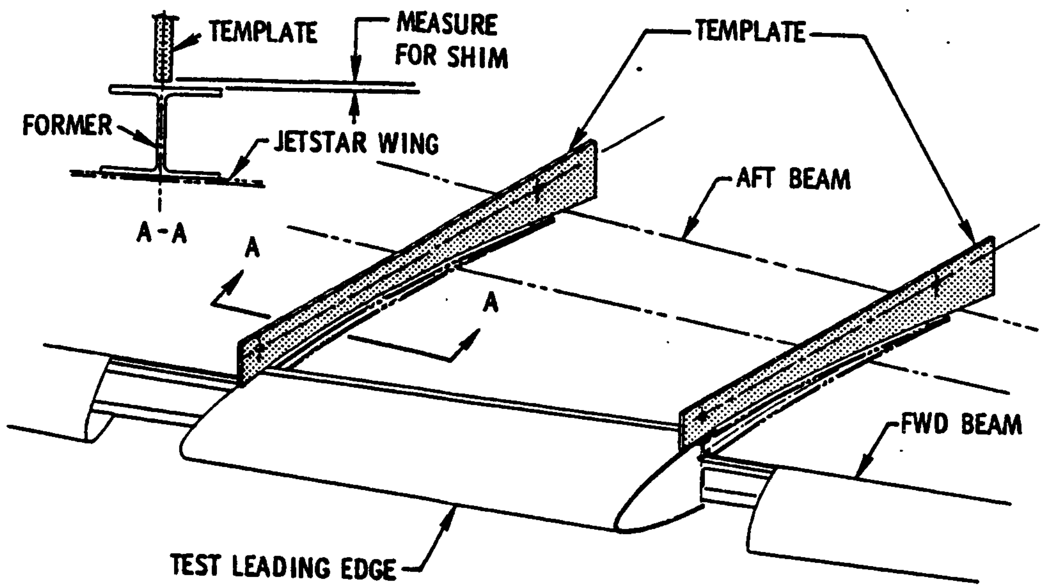

Figure 46. Over-Wing Fairing Alignment

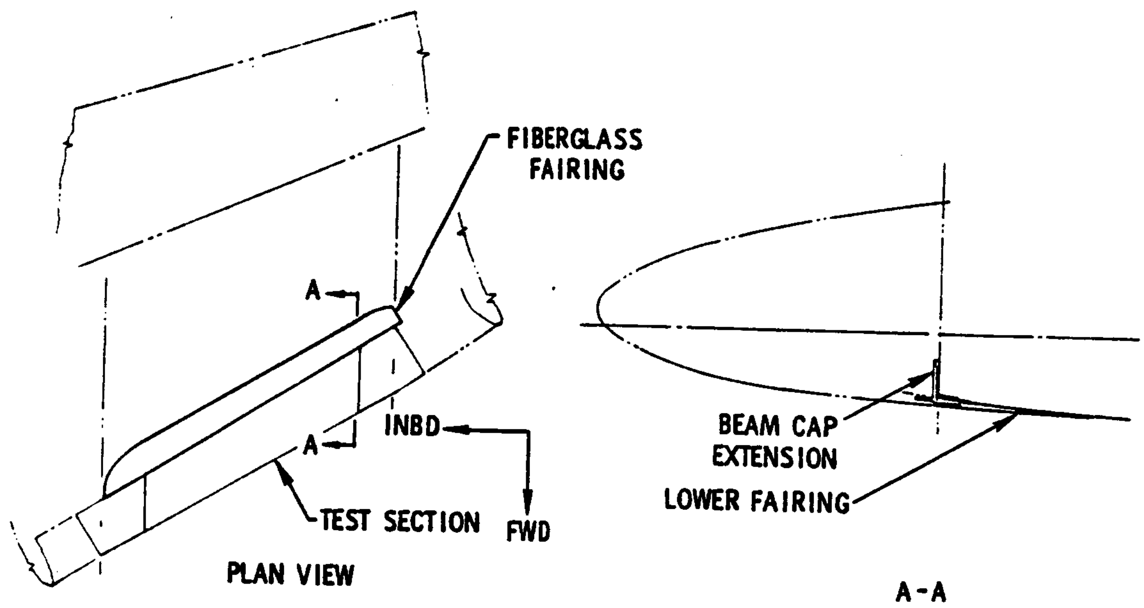

Figure 47. Under-Wing Fairing Installation

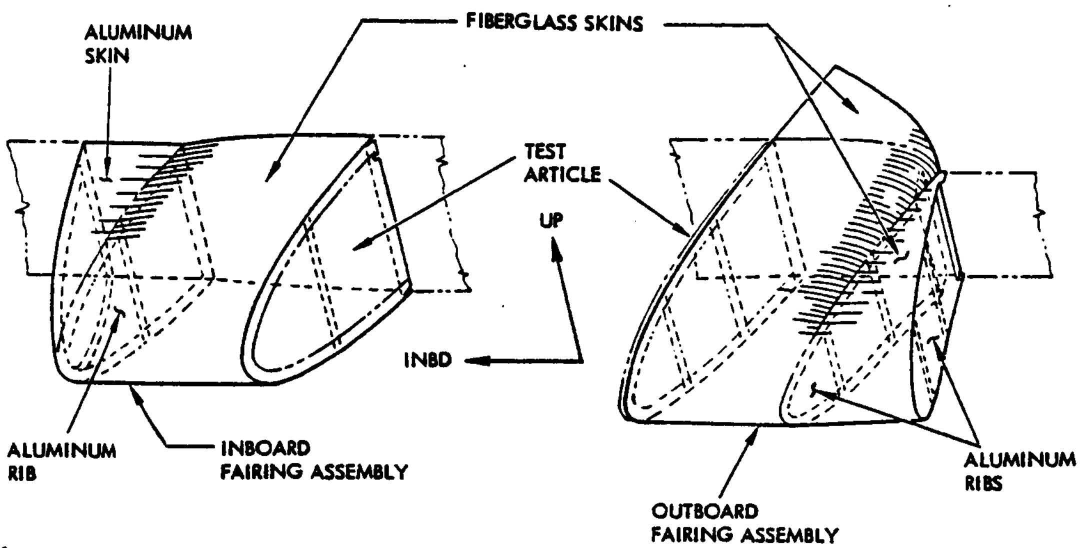

Figure 48. Test Article End Fairings

4.2 FABRICATION

A tooling and fabrication plan based on the information gained from the MD-2 test (Section 9.15) was developed.

Conventional plaster model/epoxy fiberglass tooling technology was planned with some modification to improve smoothness. The sequence of operations were quite critical due to the many operations required to produce a flight article.

4.2.1 Tooling

Templates for the leading edge were machined from loft data on a numerically controlled milling machine. Each template was inspected to contour using a coordinate measuring machine. Accuracy was good, however, but, not up to the standard desired. These templates were then placed on a tooling plate and located into position. Templates were made on 10.16 cm (4.0 in) intervals. After all templates had been adjusted, plaster was applied. Figure 49 shows the left and right bond leading-edge models partly complete. Figure 50 shows the completed models ready for pulling female plaster splashes.

Plastic on plaster plugs were then cast into the female splashes. These plugs were then checked for waves and smoothness using the Lockheed developed ripple measuring unit (RMU). Figure 51 shows the probe being used to measure the plug for the left-hand leading edge. This unit would give instantaneous readings of out-of-tolerance areas which could be smoothed and rechecked. Local low spots were marked for removal on the final tool where there would be a high spot. The RMU proved invaluable in obtaining the smoothness and wave-free surface required. Figure 52 shows the portable computer which presented the data on surface smoothness.

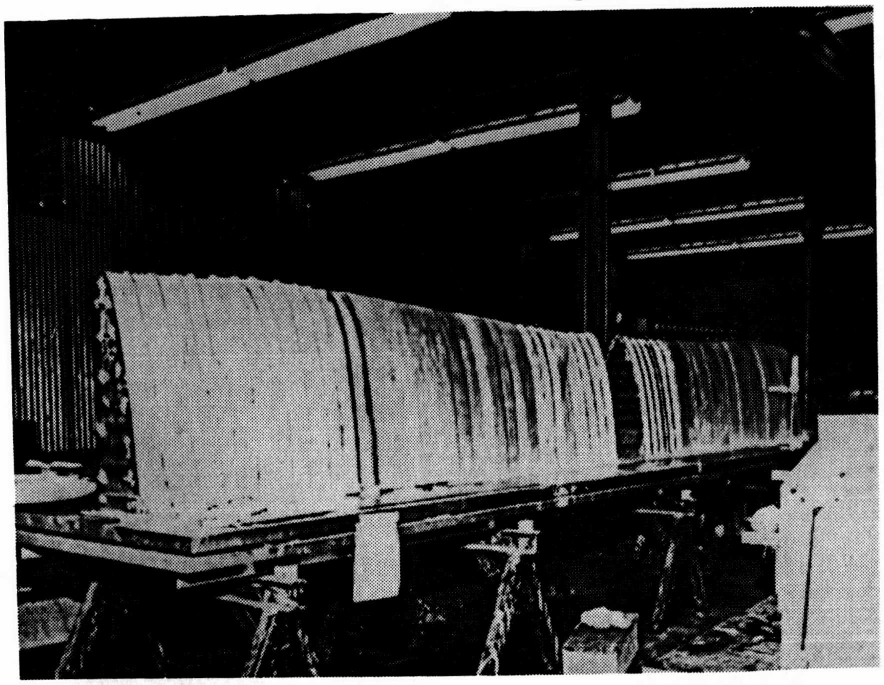

Figure 49. Leading Edge Master Models Partly Complete

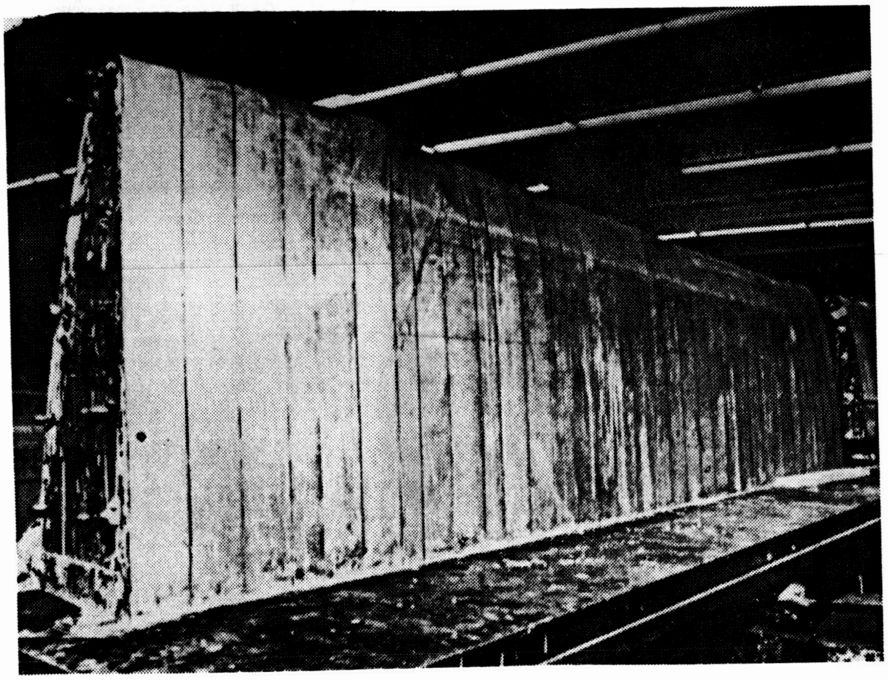

Figure 50. Completed Master Models

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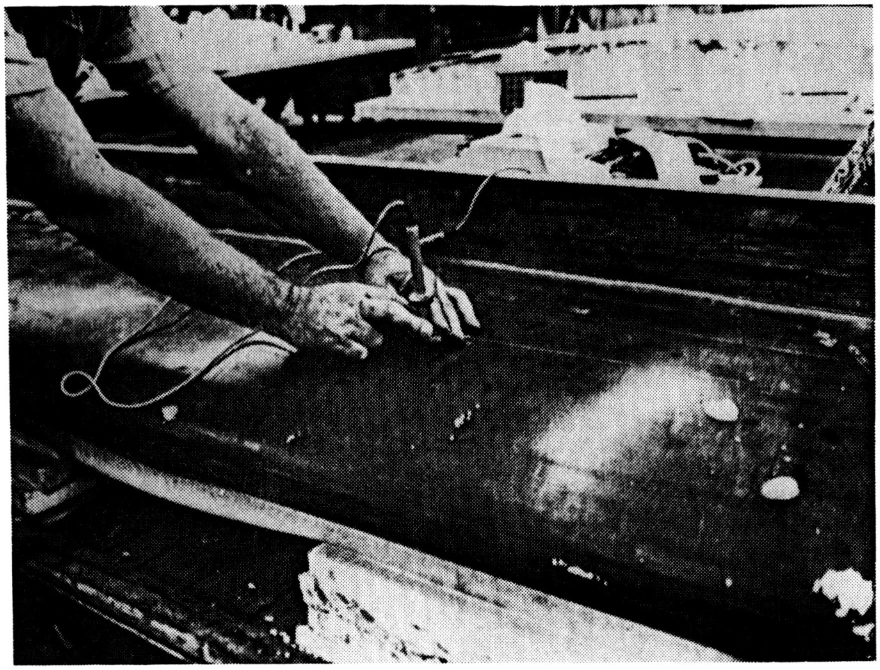

Figure 51. Ripple Measuring Unit Probe being Used to Check for Spanwise Waves on the Lefthand Leading Edge Mole Plug

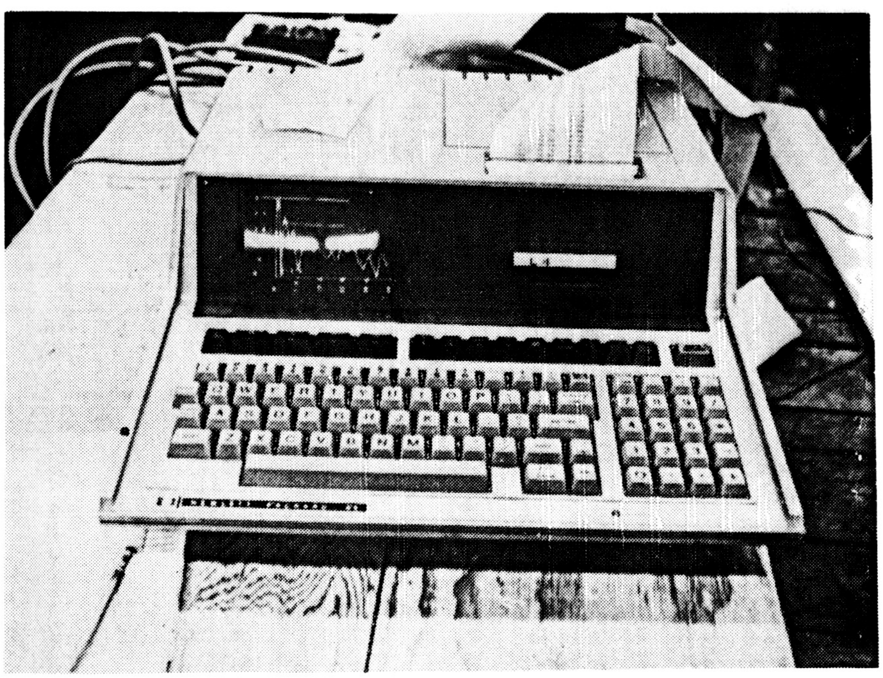

Figure 52. Computer which Produced an Instant Wave Readout

After the plugs were smoothed to within wave tolerances, a high temperature epoxy fiberglass female tool was laid up and cured on the plug. Figure 53 shows the mold for the left-hand leading edge while Figures 54 and 55 show the two piece mold for the right-hand McDonnell Douglas leading edge.

The first assembly on the tool try article was produced on this mold. As the tool encountered multiple heat cycles, movement and distortion of the mold was apparent. Since the plug was still in good condition a new tool was made with 1.27 cm (0.5 in) thick walls rather than the 0.95 cm (0.375 in) wall used on the original mold. In addition headers wre placed on 15.24 cm (6 in) centers rather than 30.48 cm (12 in) centers. Finally the mold was given a long, post cure at 190.5°C (375°F) prior to use. While this mold was more stable than the original unit, it still required rework to maintain contour and smoothness.

4.2.1.1 Fairing Tools

Fairing master models were made using the same procedures as those for the leading edge tools. A view of the models in work and a completed model are shown in Figures 56 and 57, respectively. Leading-edge end fairings were pulled from the leading-edge master as was the lower surface fairing. In all, a total of 17 fairing layup blocks were made.

4.2.1.2 Other Tools

The slot locator tool was scaled-up from the approach used on the MD2 test and is shown in Figure 58. The sloping diaphragm tool, Figure 59, was produced by machining an aluminum plate to the specified twist using a three-axis computer controlled milling machine. The sloping diaphragm tool is shown in

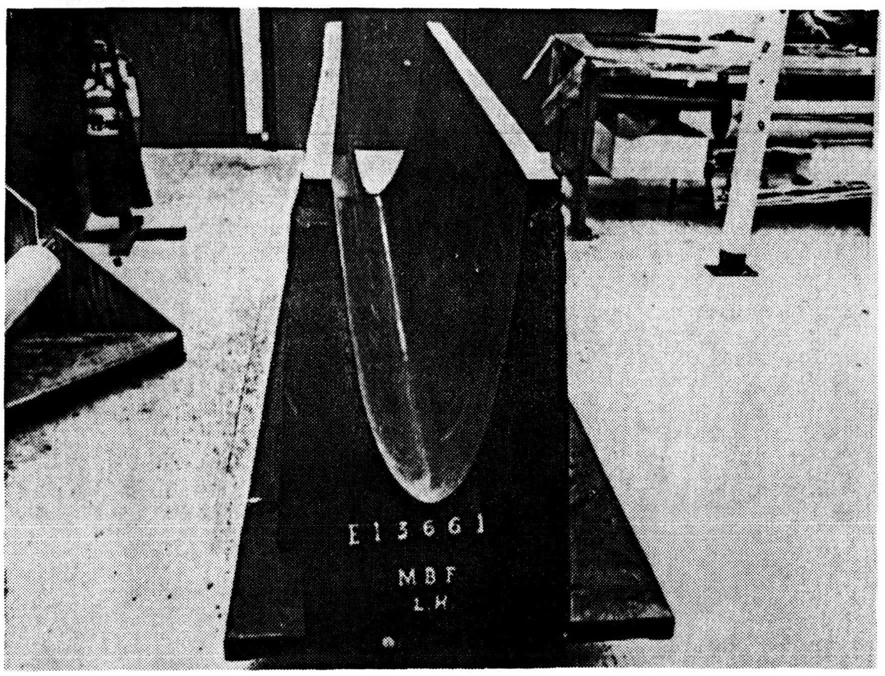

Figure 53. Left Hand Leading Edge Mold

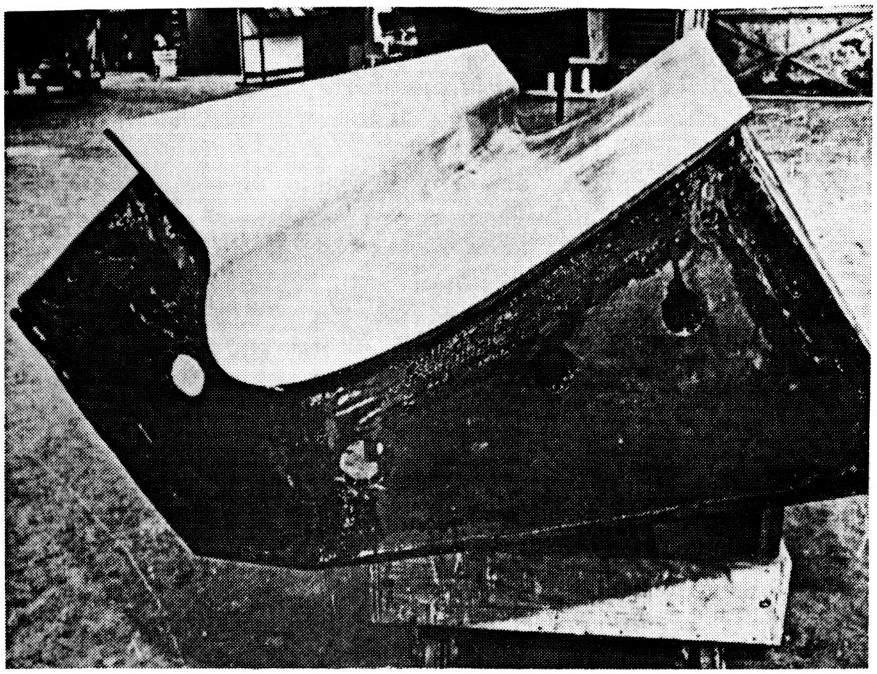

Figure 54. Right Hand Upper Surface Mold for DAC

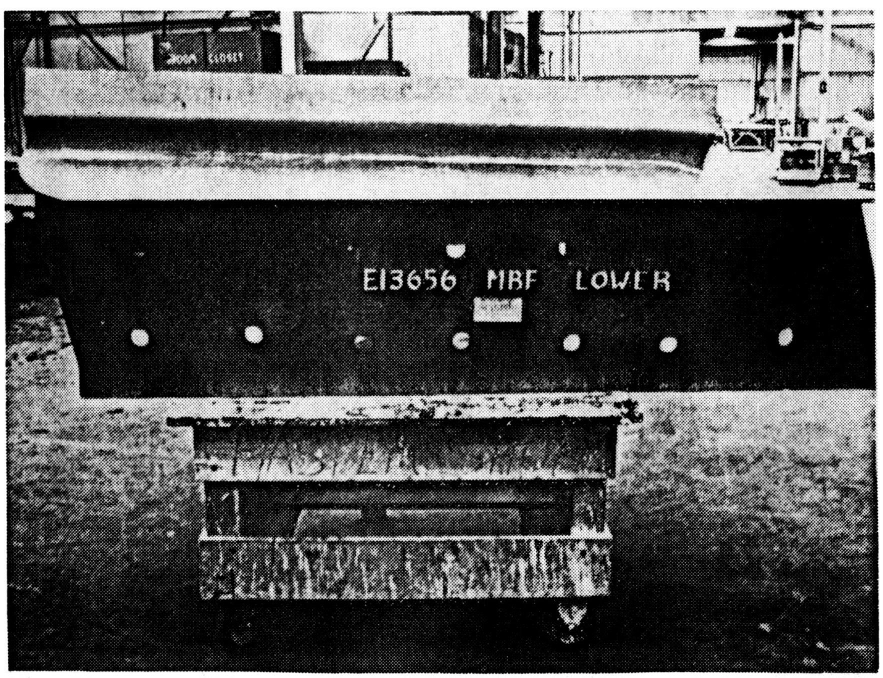

Figure 55. Right Hand Lower Surface Mold for DAC

Figure 56. Upper Surface Fairing Master Model with Templates in Place

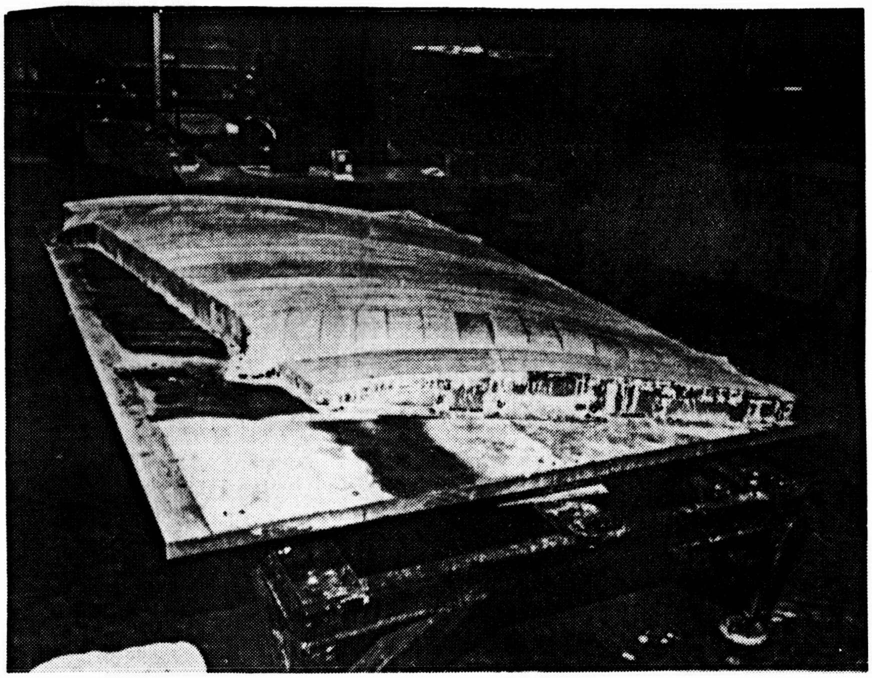

Figure 57. Completed Left Hand Upper Surface Model

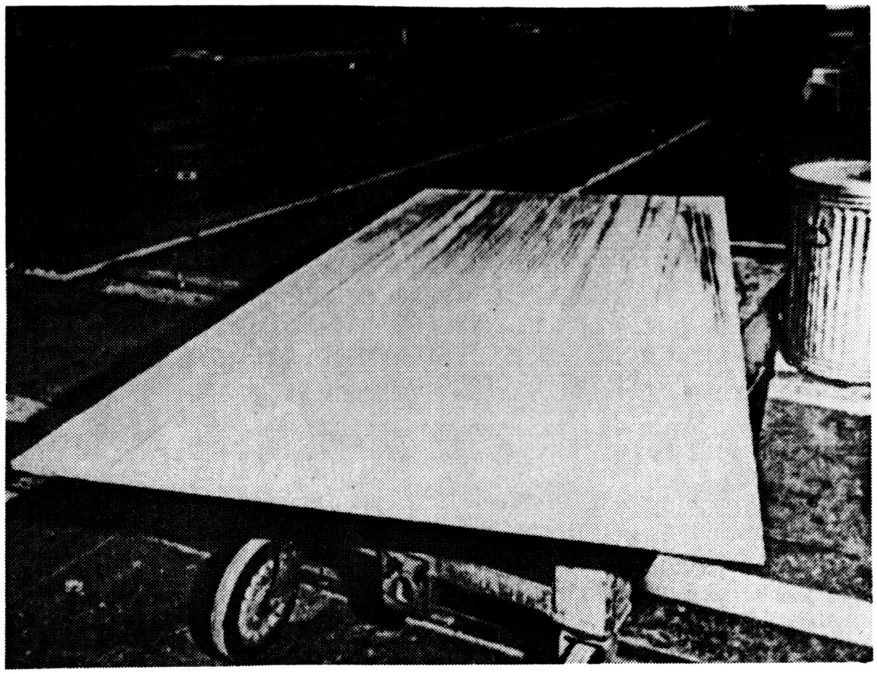

Figure 58. Slot Locator Tool

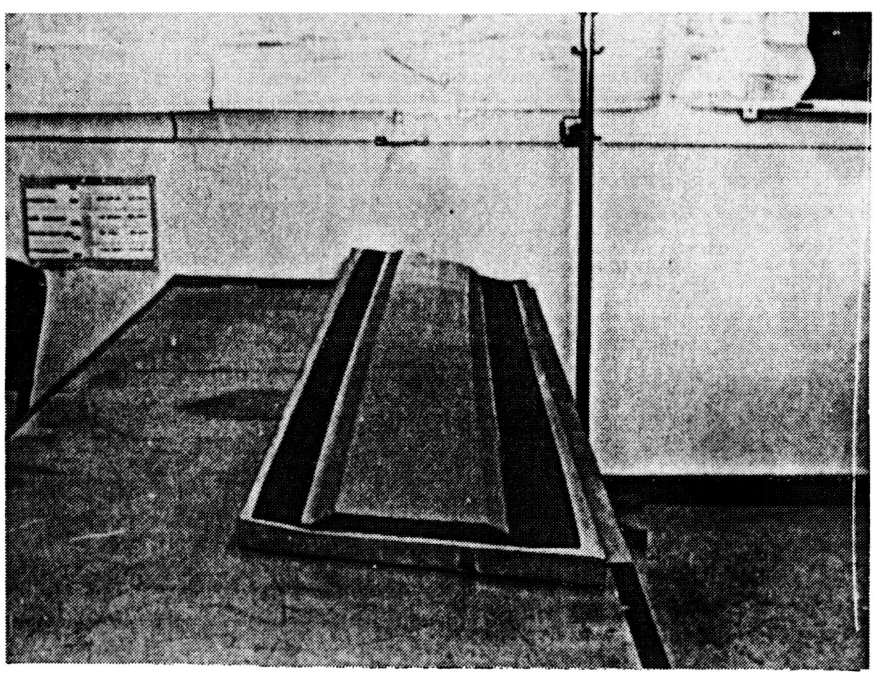

Figure 59. Sloped Diaphragm Layup Tool

Figure 59 with the assembly partially laid up. The compression molding die used to produce the slot duct inserts is shown in Figure 60. Other tools used were conventional form blocks to fabricate the various metal ribs and angles.

4.2.2 Substructure

Fabrication of the basic leading edge substructure was a step-to-step operation as demonstrated in the MD-2 test. The peel ply subassembly was laid up on the slot duct locator tool and cured. Slot duct inserts had previously been fabricated using the compression molding process described in MD-2. Figure 61 shows the first assembly used in the tool try article. Some problems were encountered with the inserts trying to peel from the surface prematurely. This problem was solved by rebonding prior to proceeding with the fabrication. No problem was experienced with this during the flight article fabrication. The peel ply subassembly was trimmed to 2.54 cm (1.0 in) oversize on all edges and a layer of American Cyanamid FM400 adhesive applied. Four plies of graphite fabric prepreg were then applied between the slot ducts to prevent bridging of the continuous plies of the outer skin. Figure 62 shows the trimmed subassembly with adhesive in place and filler plies being added.

The first ply of the outer skin was then laid up, and the assembly inserted in the leading-edge mold. This subassembly had tooling holes to coordinate to the proper position in the mold. Figure 63 shows the outer skin in the mold. This skin was cured at 586 KPa (85 psi) to develop the highest density outer skin possible. The next step was to install the collector ducts as shown in Figure 64. After all collector ducts had been installed, core splice adhesive was applied to all edges of the ducts and the honeycomb fitted. The assembly was then bagged and autoclave cured. This allowed the core to be sanded smooth as required. Also a detail inspection could be made as to position and straightness of the collector ducts. The substructure assembly at this stage is

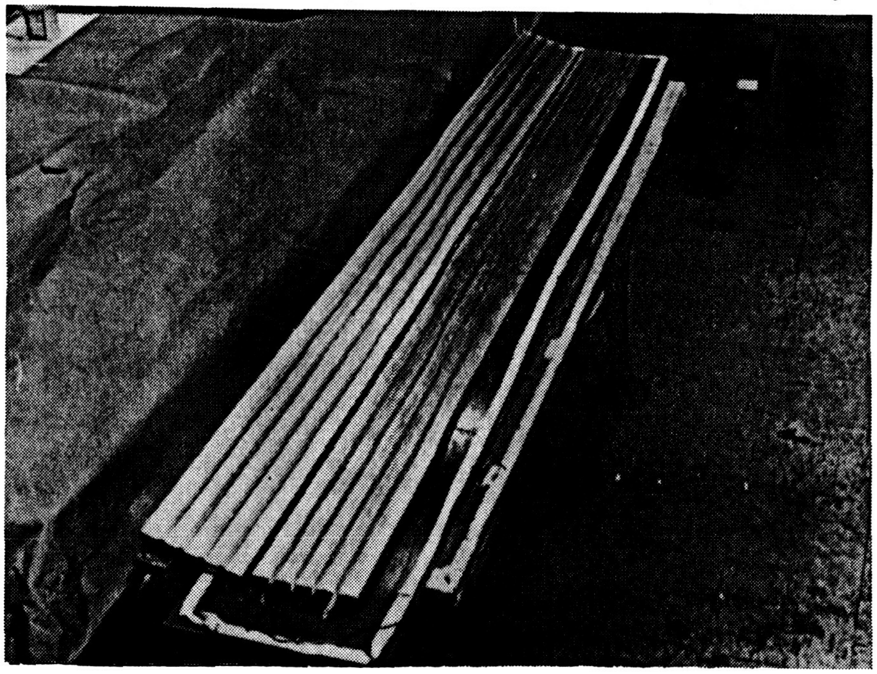

Figure 60. Slot Duct Insert Die

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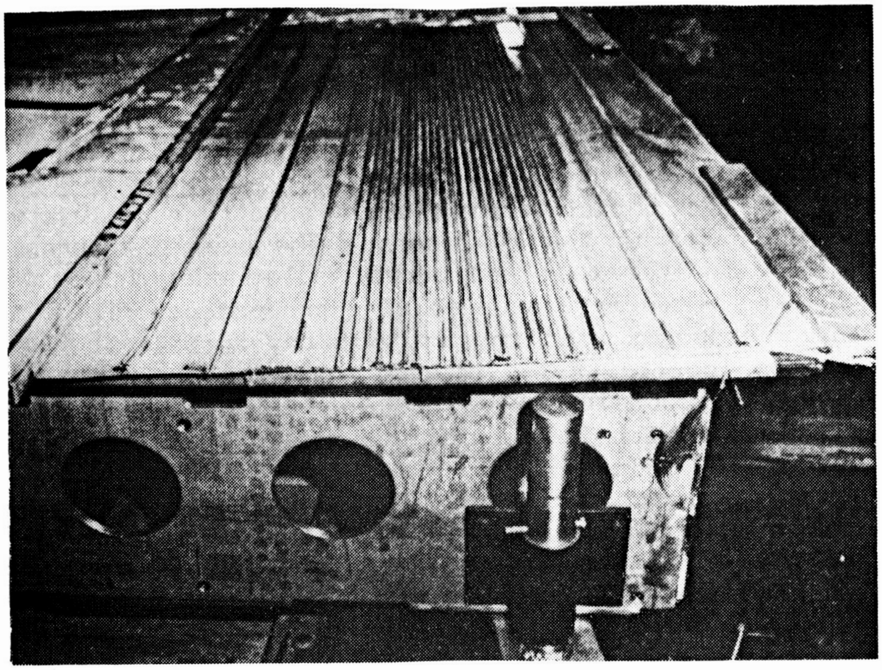

Figure 61. Peel Ply Subassembly for Tool Try Article

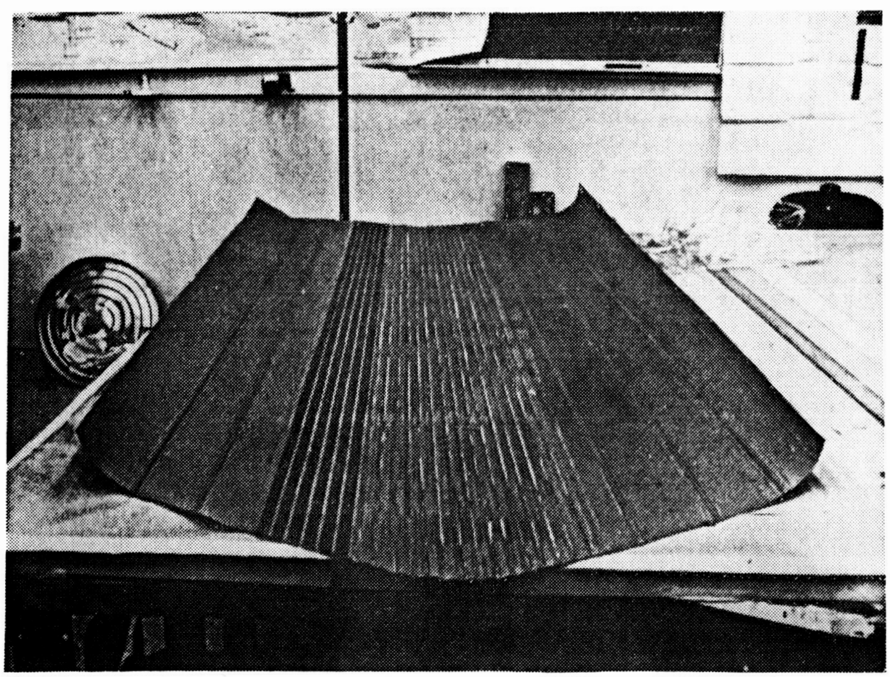

Figure 62. Peel Ply Subassembly with Adhesive Applied and Filler Stacks being Applied

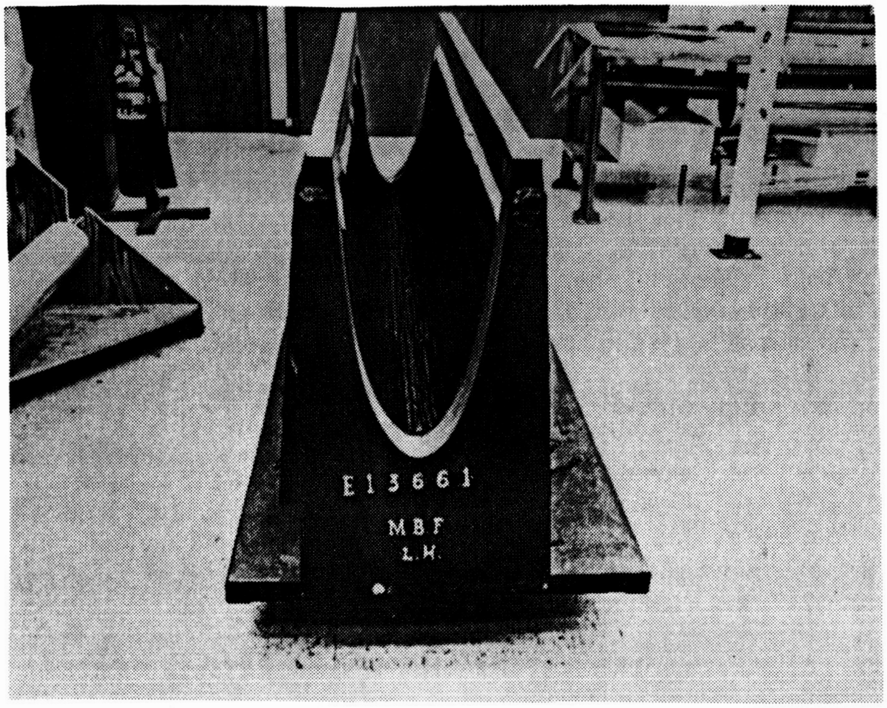

Figure 63. Outer Skin in Leading Edge Mold

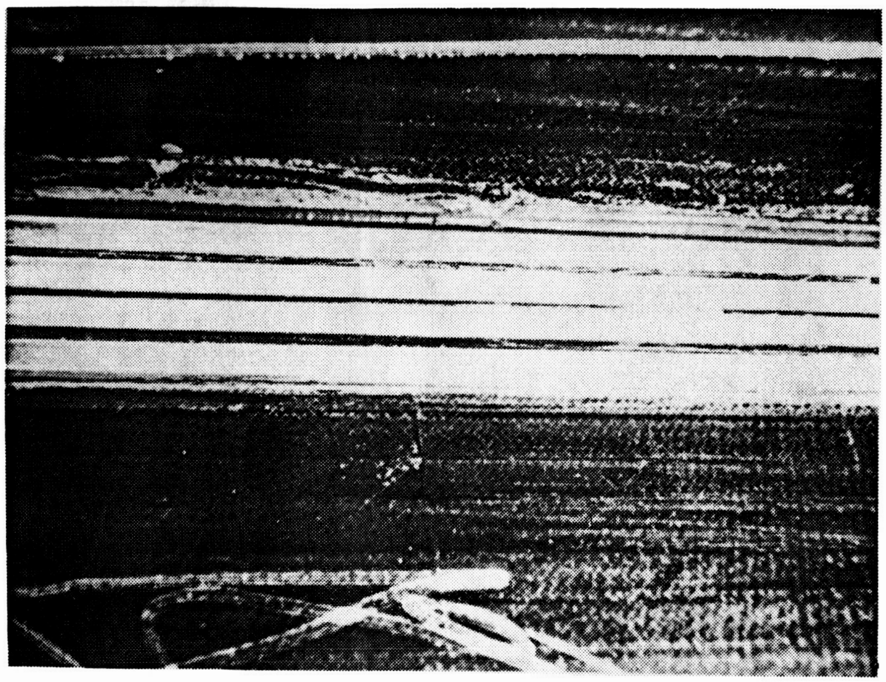

Figure 64. Collector Ducts being Bonded into the Cured Outer Skin Assembly

shown in Figure 65. An inner graphite skin was then laid up and cured at 241.3 KPa (35 psi) which was the maximum pressure allowable for the core. Figure 66 shows the assembly at this point. Bars had been added to the tool to correct an out-of-contour condition.

Installation of the upper and lower attach angles by bolting and bonding was the next step and is shown in Figure 67. Dummy upper spar and spar cap extensions were machined and attached to the 12 percent chord line of the assembly. This allowed the assembly to be mounted to a dummy front spar which would support the unit during drilling of metering holes, slotting, testing and shipment. Removal of the peel ply from the outer surface was accomplished at this point. The tool try article exhibited a good surface, requiring only minor filling and cleanup before drilling the metering holes.

There are approximately 7900 metering holes from the slot ducts into the collector ducts in each assembly. These were drilled with #69 0.074 cm (.029 in) solid carbide drills using semiautomatic equipment as shown in Figure 68. The system worked extremely well. Hole diameter was consistently between 0.071 and 0.074 cm (.028 and .029 in) in diameter.

Removal of the peel ply from the flight article surface revealed voids in the outer surface. In addition, the center leading edge apparently had floated away from the tool producing a swag along the zero percent line. This is shown rather vividly in Figure 69 and Figure 70. First, attempts were made to locally fill the voids with EA934, a high strength, dense paste adhesive, and then co-ordinate to the mold. This was not completely successful. Finally, the entire surface was splined with EA934 adhesive and sanded to fit the mold. At the same time, some high spots were removed from the tool.

New slot ducts were then cut in the outer surface using a 1.57 cm (0.62 in) radius wheel cutter. This procedure produced a perfect slot duct which was of

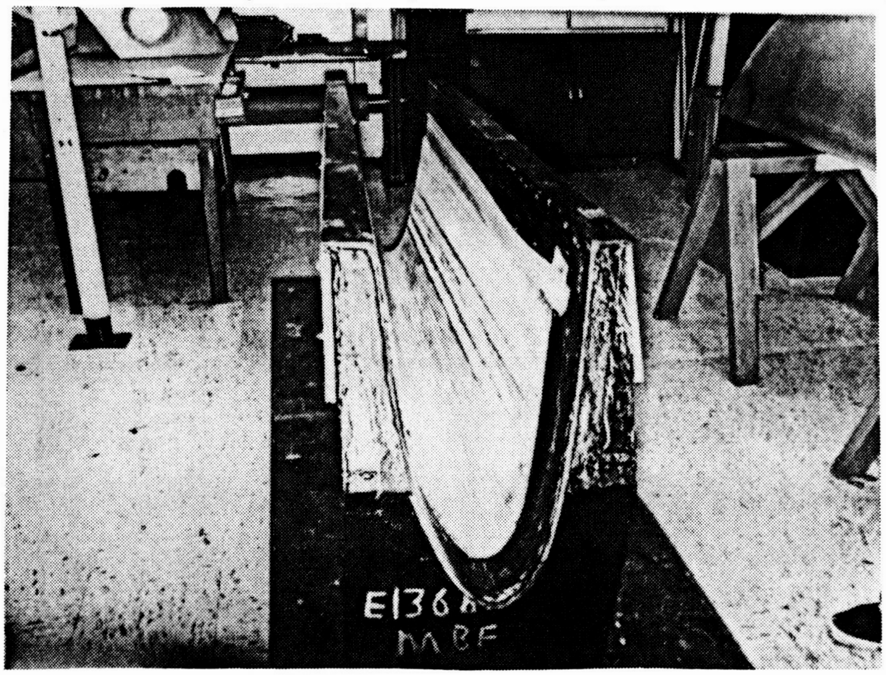

Figure 65. Substructure with all Core and Ducts Bonded in Place

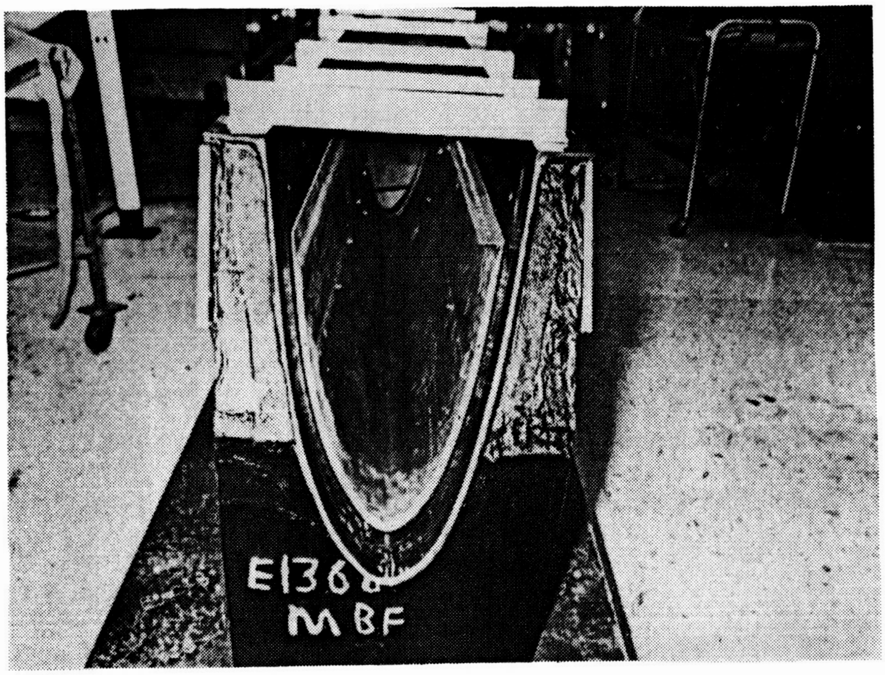

Figure 66. Substructure Assembly with Outer Skin in Place and Cured. Holes for Attaching the Upper and Lower Attach Angles had also been Drilled

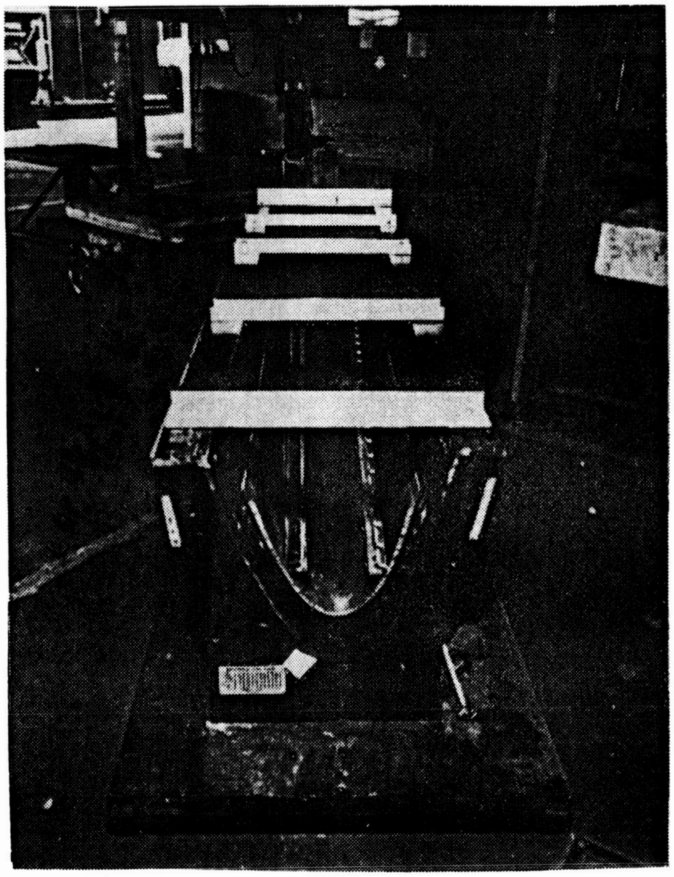

Figure 67. Assembly with Upper and Lower Attach Angles Installed

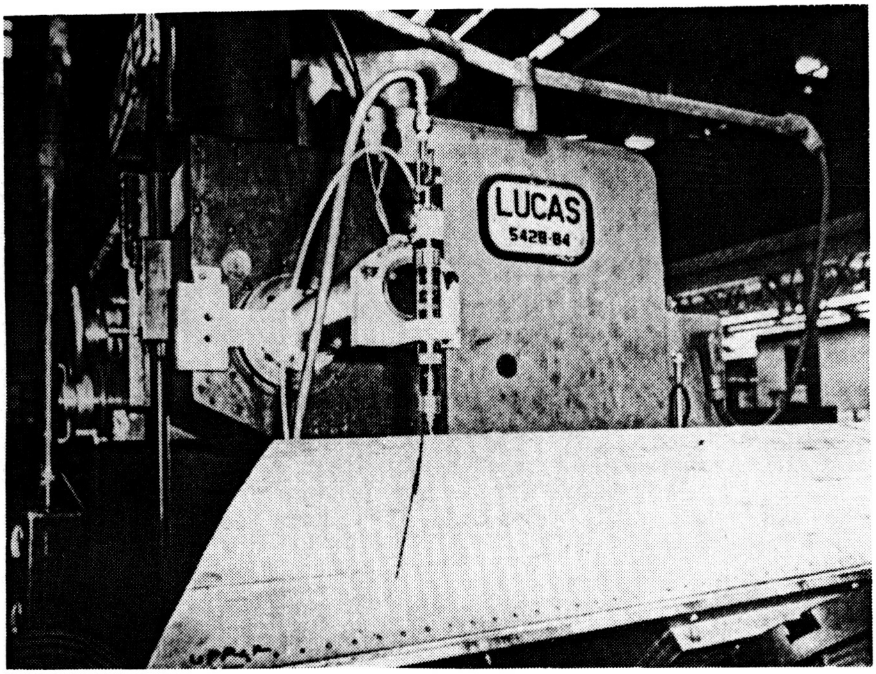

Figure 68. Drilling Metering Holes with Semi-Automatic Drilling Equipment

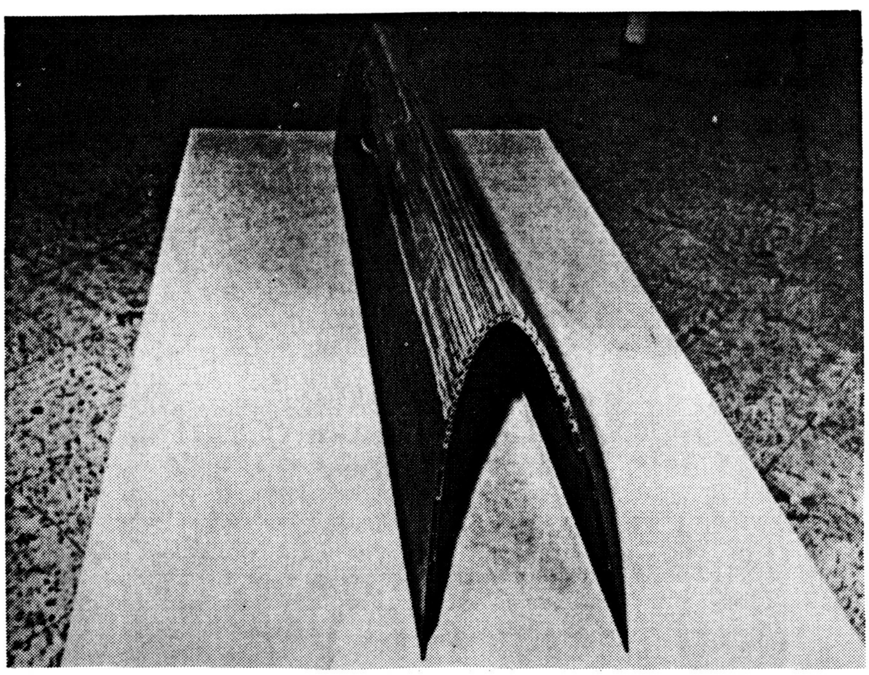

Figure 69. Flight Article Subassembly with Peel Ply Partially Removed

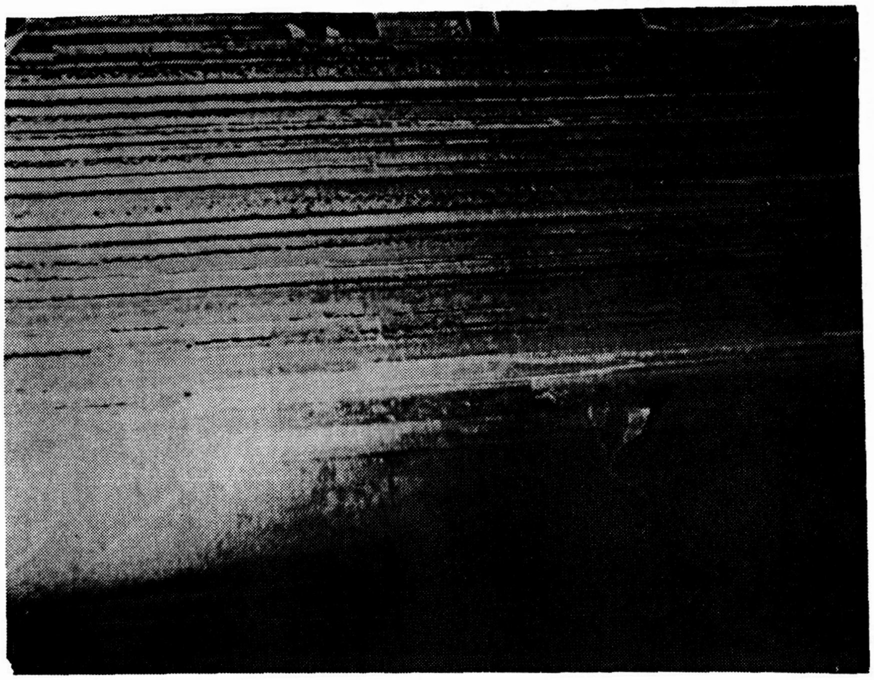

Figure 70. Closeup View of Outer Surface Voids Around Slot Ducts

more consistant shape and straightener than that previously obtained. Figure 71 shows slot ducts being cut into the substructure and the completed substructure is shown in Figure 72. Figure 73 is a closeup view of the machined slot ducts with metering holes drilled. Each slot was lined up directly over the collector duct and as close to the design location as possible. After cutting the slot duct, the cutter was removed and replaced with the drilling unit. This allowed metering holes to be drilled in the exact center of the slot duct.

Other details were produced concurrent with the leading edge substructure. Figure 74 shows the molded composite upper attach angle. The composite faced honeycomb sandwich sloped diagram is shown in Figure 75. Figure 76 shows a representative assembly of the many details which the system would require. The outboard fiberglass leading edge fairings are shown in Figure 77.

4.2.3 Skin Forming

The laminated skin looked very promising in the MD-2 test with no apparent initial closure on straightening. During the suction tube attachment study the article was heated to 121.1°C (250°F) to cure the adhesive used to bond the suction tubes to the inner skin of the substructure. The unit was not restrained in the tool at the time of heating. Following the heat cycle it was observed that some straightening of the titanium skin had occurred. This was attributed to heating while unrestrained and a precaution of always placing the unit in the tool during any heating was incorporated in the manufacturing plan.

A bonded skin for the tool try article was prepared and bonded to the substructure at 93.3°C (200°F) and 241.3 KPa (35 psi) in the autoclave. A verification bond cycle was conducted prior to bonding and gave good results. Fit between the skin and substructure was good since they were both produced on the same tool.

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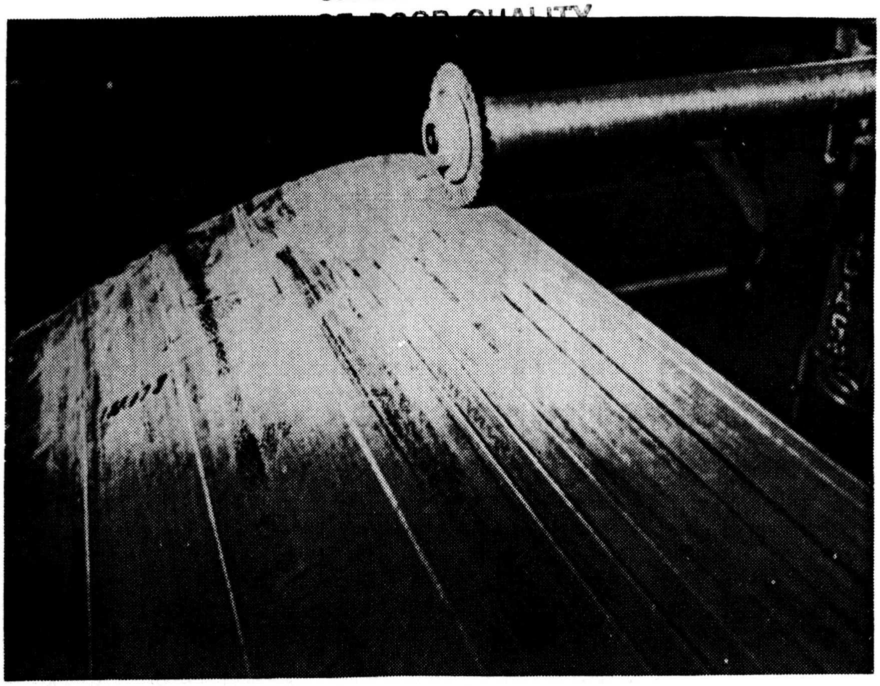

Figure 71. Cutting Slot Ducts in Flight Article with Carbide Wheel Cutter

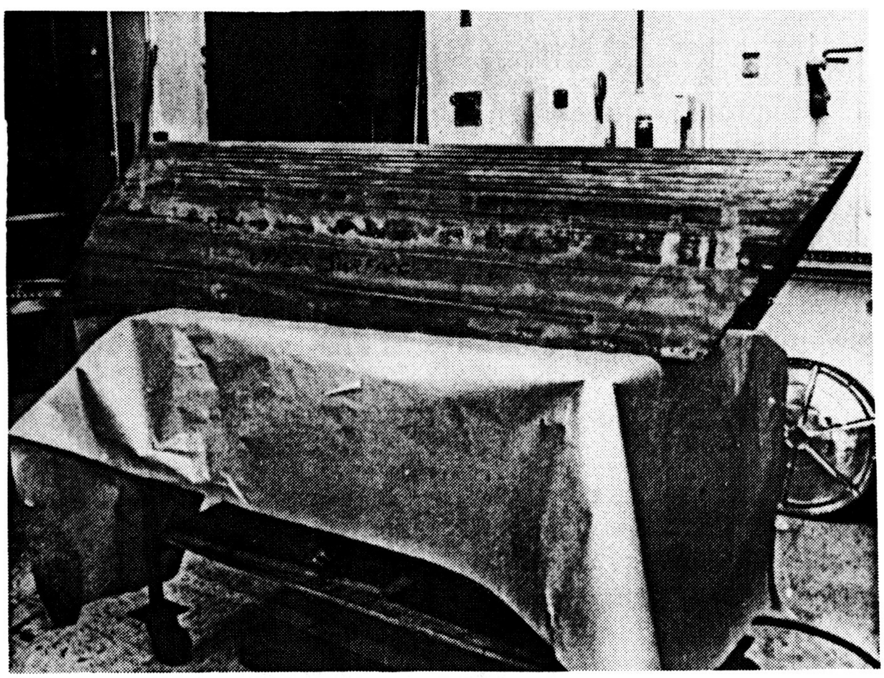

Figure 72. Completed Flight Article Substructure

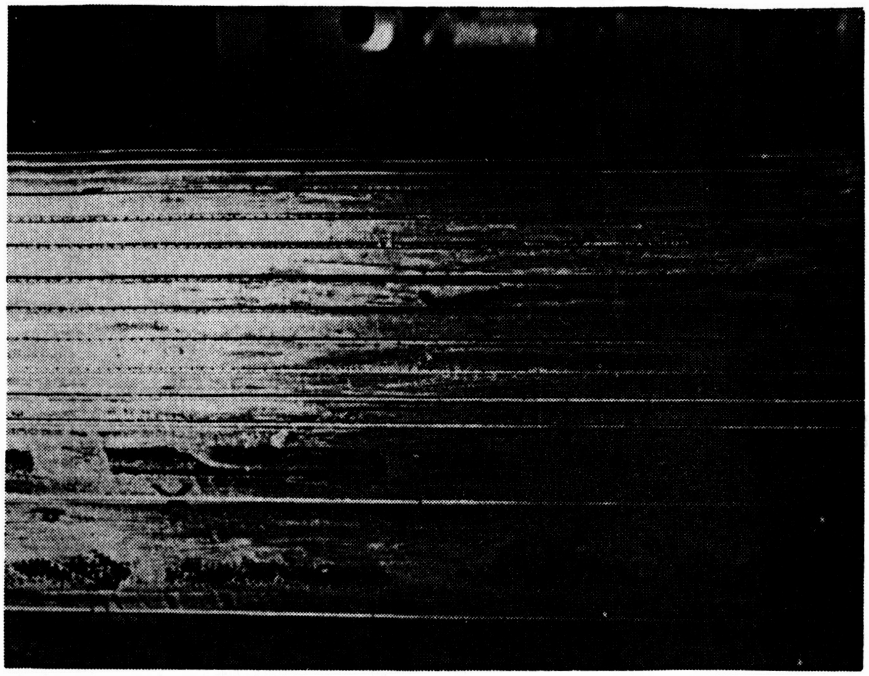

Figure 73. Closeup View of Flight Article After Metering Holes have been Drilled in the Machined Slot Ducts

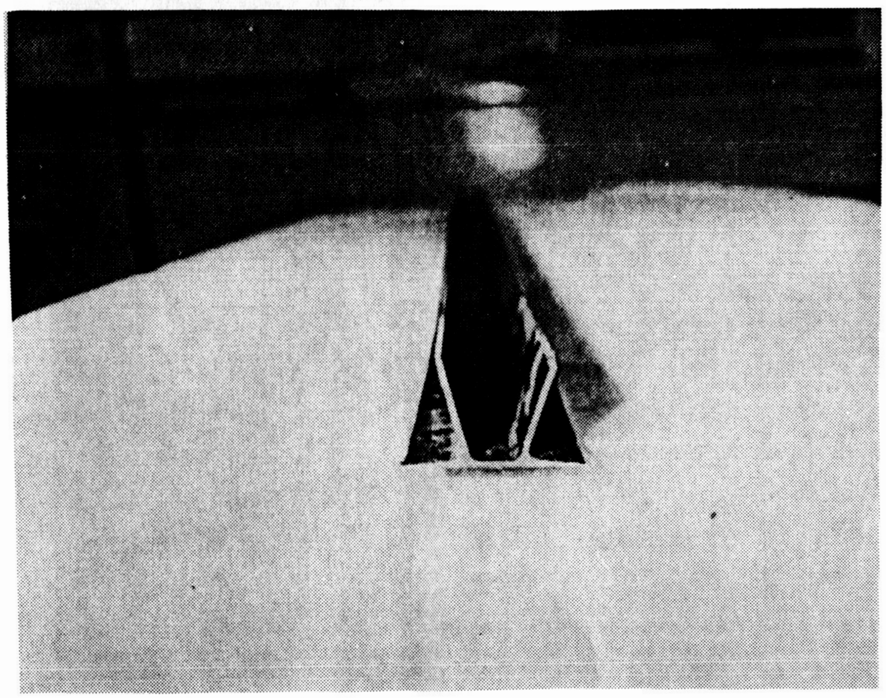

Figure 74. Molded Composite Upper Attach Angle

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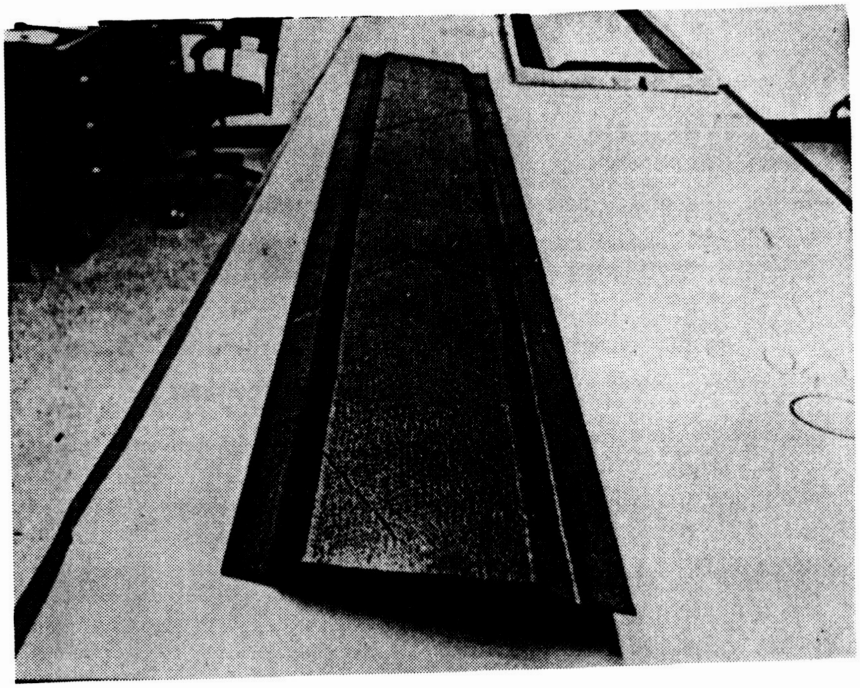

Figure 75. Sloped Diaphragm

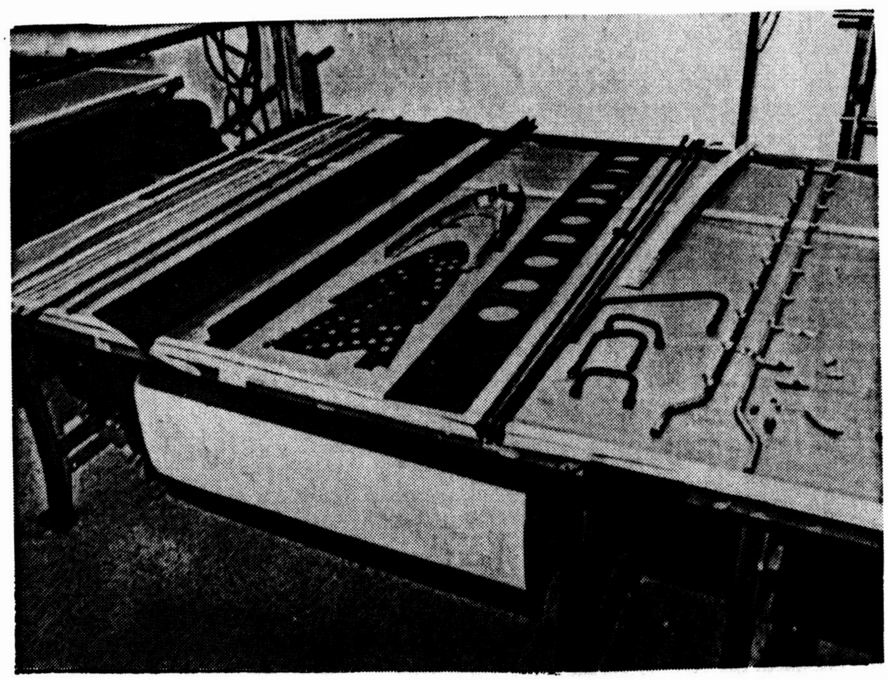

Figure 76. Some Representative Hardware Details in Addition to the Leading Edge Substructure

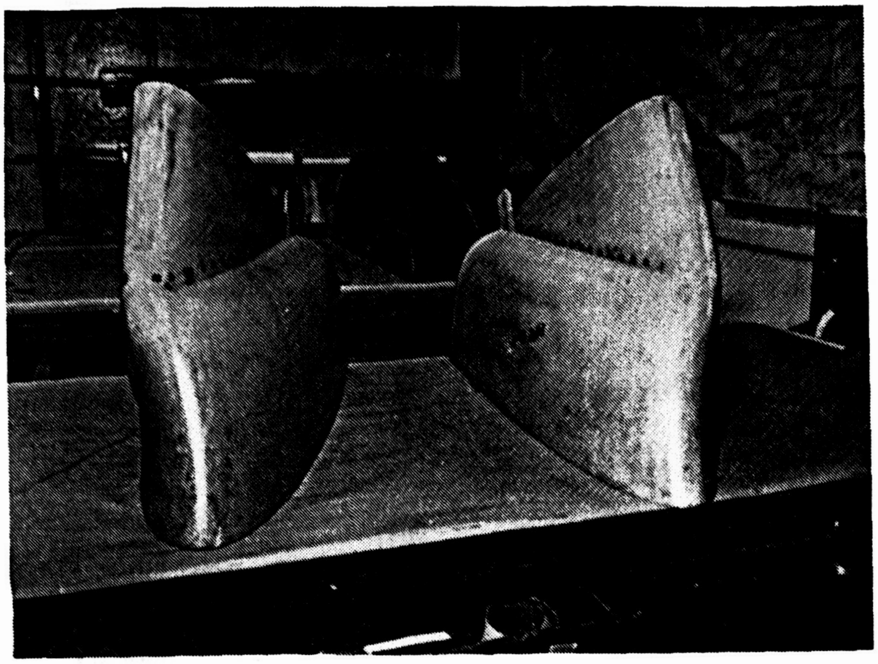

Figure 77. Outboard Leading Edge Fairings

Following slotting it soon became apparent that a relaxation of the adhesive was taking place allowing the titanium to form a series of flats between slots in the area of greatest stress. It was decided that a hot formed titanium skin was probably the only solution to the slot opening-closure problem. Lockheed's Manufacturing Research organization had considerable experience in making integrally heated cast silica tools. They originally planned to make a male and female matched set, but personnel at NASA Langley suggested that a single male mold would probably work better. This was cast directly in the layup mold with integral nichrome heaters. It was planned to drop form the 0.051 cm (.020 in) thick titanium sheet by lightly pressing over the heated plug. Stainless steel slip sheets were to be used on both sides of the titanium. Heavy insulation blankets covered the part. Forming was to be accomplished at 732° - $760^{\circ}C$ (1350° - $1400^{\circ}F$).

The tool try article had been cut into two pieces, an inboard section which had been tested as the static test article and the outboard section which was to be the sonic test article. Static testing had been accomplished prior to the decision to form a new skin. It was decided that the sonic test article would be reskinned with a hot formed skin.

The first skin formed was cut to fit the sonic test article which had been deskinned. Figure 78 shows the outboard end of the hot formed skin and the deskinned sonic test article with fresh adhesive applied ready for bonding. After bonding an inspection revealed voids in the nose area. Further investigation revealed that the skin was not being formed to contour at the nose due to the stainless slip sheets holding the skin off the tool. Another problem was caused by the skin "straight" lining over small waves.

The cast silica tool was then reworked to give a straight line at any point along the nose. Several skins were formed using different techniques; however,

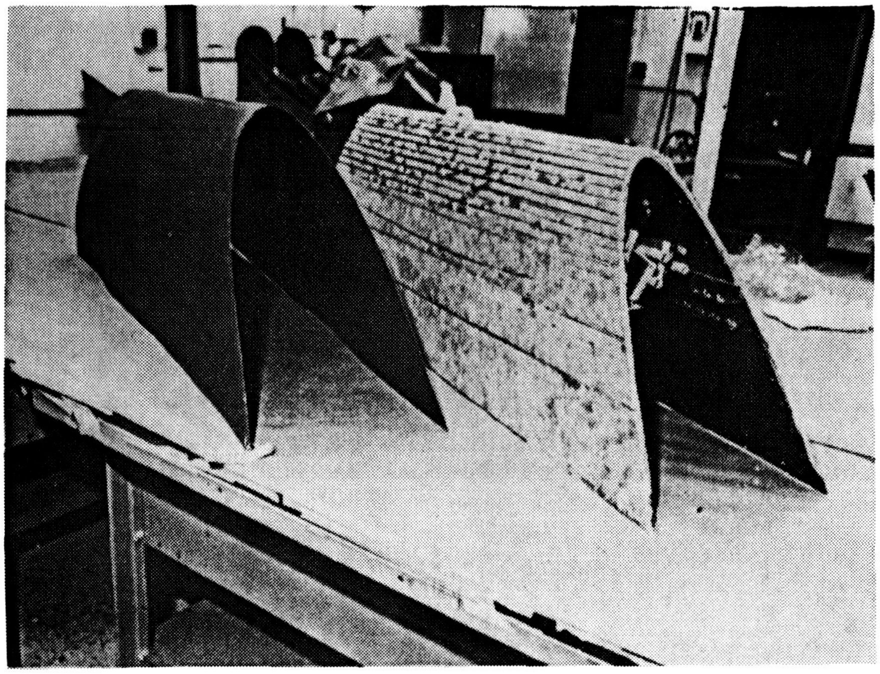

Figure 78. Sonic Test Article and Hot Formed Titanium Skin Ready for Bonding

they were unacceptable in that the nose never exactly fit to contour while at the same time eliminating "oil cans" along the 12 line percent, Figure 79. The cast silica tool was not capable of maintaining a sufficiently constant temperature to form a skin without oil cans. A larger, more precisely heated tool would have formed the titanium. Likewise it could probably have been accomplished in a furnace. Time and budget contraints prevented trying either of these approaches. It was noted however, that the hot formed skin was stable. There was no movement on the sonic test article either before or after the sonic fatigue test.

It was proposed by the NASA Technical Monitor that roll forming followed by stress relieving might be a solution. This was tried on a small piece. Roll forming the tapered section was accomplished quite easily. It was then stress relieved 648.9°C (1200°F) over a simple pipe hold fixture and no movement took place. This was followed by a full size skin. It was roll formed and stress relieved. A second local forming operation was then done to improve fit and it was stress relieved. The fit was seemingly perfect. This skin is shown in Figure 80.

4.2.4 Final Assembly

As discussed earlier, the upper and lower attach angles were installed in order that the diaphragms could be installed prior to cutting slot ducts and drilling metering holes. While they may not have been necessary, an added degree of stabilization was achieved. Following drilling of the metering holes, the substructure was prepared for bonding.

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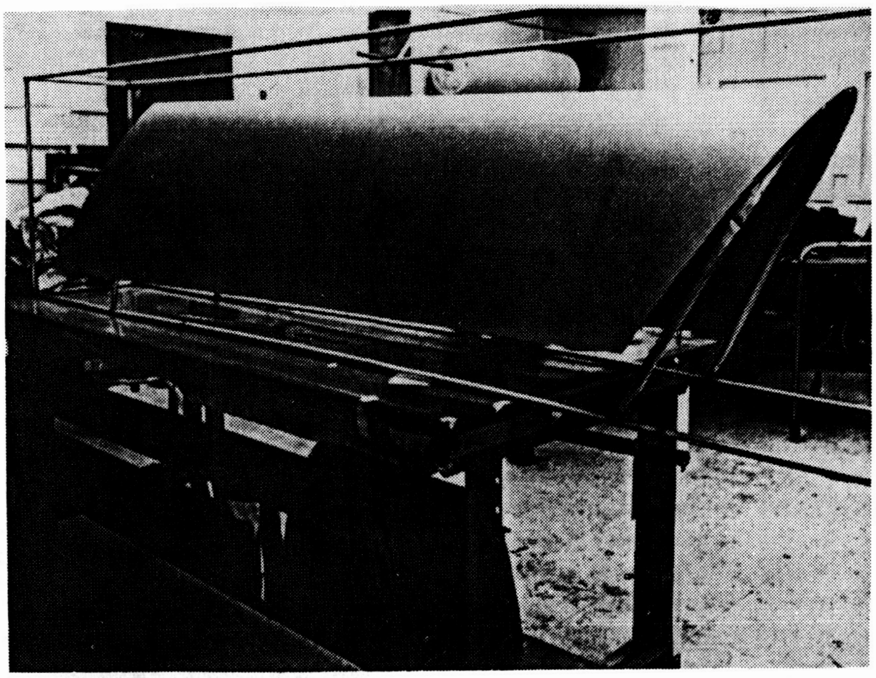

Figure 79. Hot Formed Skin - Oil Cans at and Forward of the 12% Line Caused Rejection

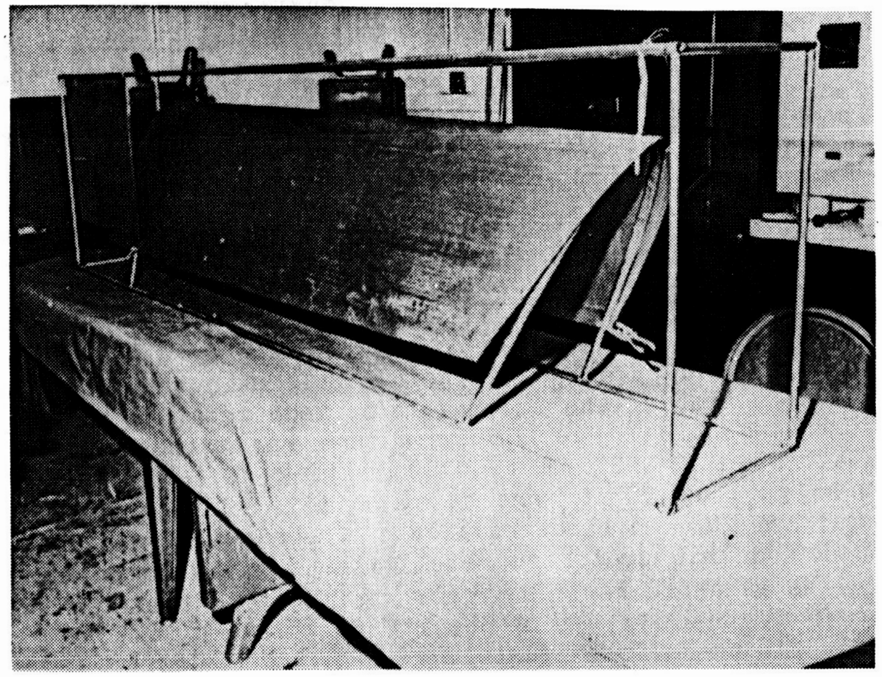

Figure 80. Roll Formed and Annealled Skin used on Flight Article

4.2.4.1 Bonding

The substructure was prepared for bonding by lightly sanding followed by a wipe with CP grade methyl ethyl ketone. Titanium cleaning was accomplished in accordance with STP 57-004. Following cleaning the skin was primed with American Cyanamid BR127 primer within four hours. As noted earlier both FM123-2 and FM123-4 adhesive was used early in the program. Chemically they are identical, the -2 variety has a mat carrier and the -4 has a knit carrier. Higher peel strength is generally obtained with the -2, so it was used on the first article.

Strips of adhesive were cut to the exact width of the land between the slot ducts and applied, followed by a very thin polyester mat positioning film. This was to serve two purposes, reduce adhesive flow and allow the formed skin to be positioned over the substructure without sticking to the normally tacky adhesive surface and causing the adhesive to slide off the land. The assembly was then placed in the tool, bagged and autoclave cured at 933°C (200°F) for 3 hours at a pressure of 241.3 KPa (35 psi).

A coin tap inspection revealed no delaminated areas in the tool try article. As noted earlier, after slotting the skin tried to return to its former shape by forming slight flats between slots and a lip at each slot. The first hot formed skin was scheduled for the sonic fatigue article. This unit was partly plumbed by this time which presented a problem in bagging to apply bond pressure. Some damage did occur during bonding which required repairs. In addition voids were present at the nose due to the skin not being formed perfectly. Figures 81 and 82 show the sonic fatigue article being repaired.

Another problem came to light when the old skin was removed from the sonic fatigue test article. The adhesive had flowed and partially filled the slot ducts. After a detail investigation it was determined that the FM123-2 adhesive flowed more than the -4 formerly used. Several tests were conducted and it was demonstrated that adhesive flow into the slot ducts did not occur when the FM123-4 adhesive was used and cured in the selected manner. This adhesive was used on all subsequent tests.

It was quite apparent that the flight article skin had to fit the substructure perfectly and a test of the fit had to be made prior to bonding. This test was achieved by conducting a verification bond run using a layer of adhesive between two layers of 0.0025 cm (.001 in) thick Teflon film. The part was bagged and cured just as if it were being bonded. After curing, the adhesive film was removed and gave an exact replica of the adhesive as it would be in the assembly. Thick and thin spots are revealed as are any potential voids or unbonded areas. The adhesive replica from a verification cycle on the last formed skin is shown in Figure 83. As a result of this test the skin was rejected since it did not fit the part adequately.

During the verification cycle on the final roll formed skin, excessive pressure and a fast heat up cycle was used in error by the manufacturing shop which caused breakage of the Teflon film over many metering holes and adhesive flow into the metering holes. These had to be drilled out as well as the collector ducts. Figure 84 shows a closeup view of the clogged metering holes. Figure 85 shows the fourth and final verification bond replica while Figure 86 shows the flight article with adhesive applied ready for bonding.

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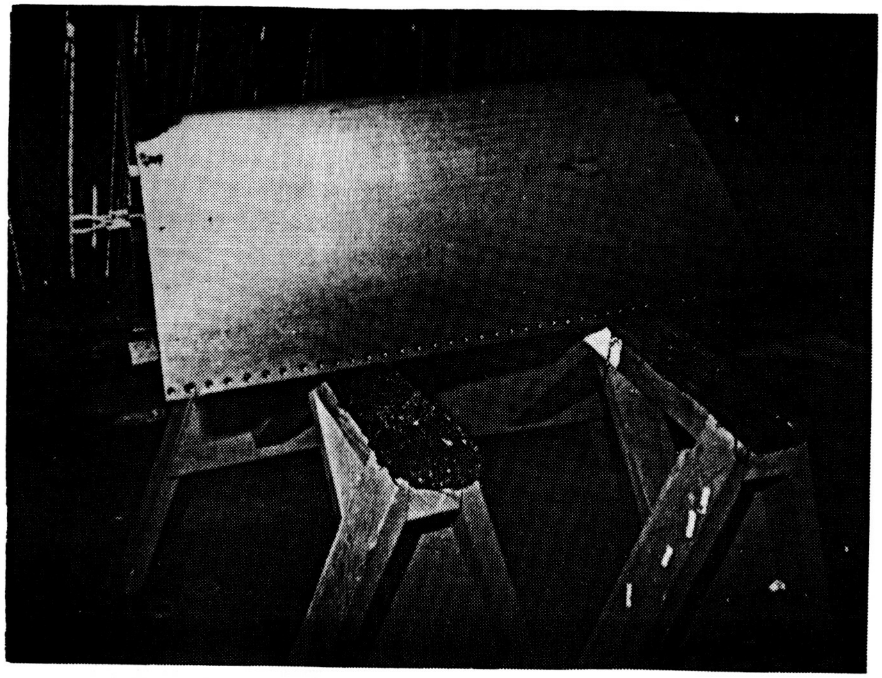

Figure 81. Sonic Fatigue Article Reskinned with Hot Formed Titanium Skin

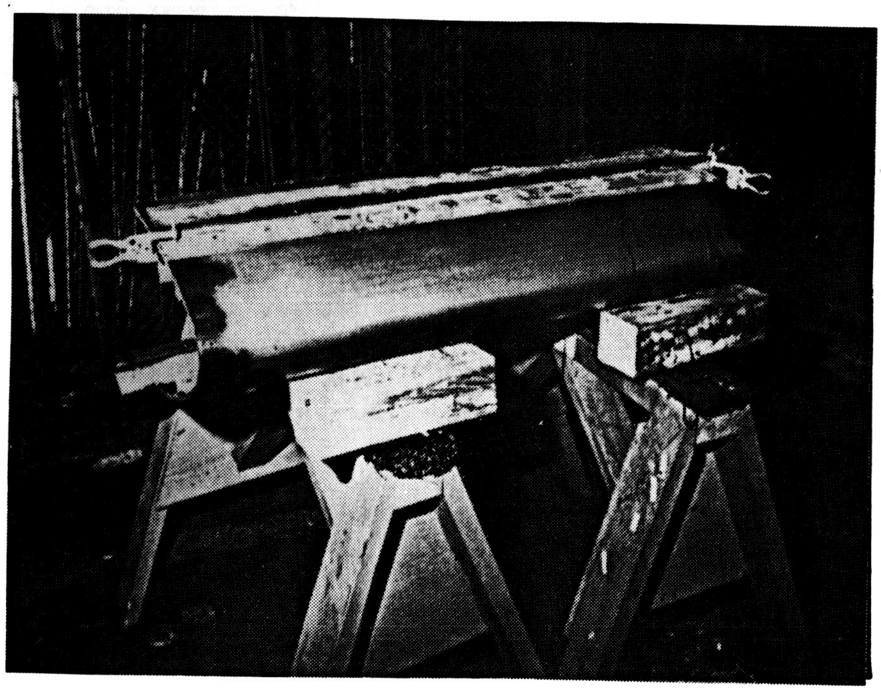

Figure 82. Sonic Fatigue Article Being Repaired

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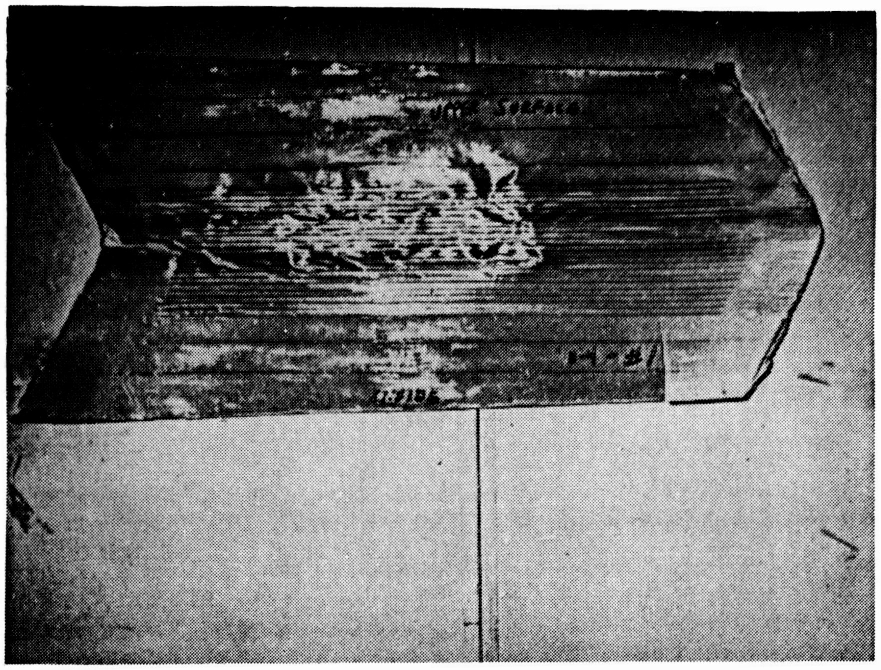

Figure 83. First Verification Bond Cycle with Last Hot Formed Skin

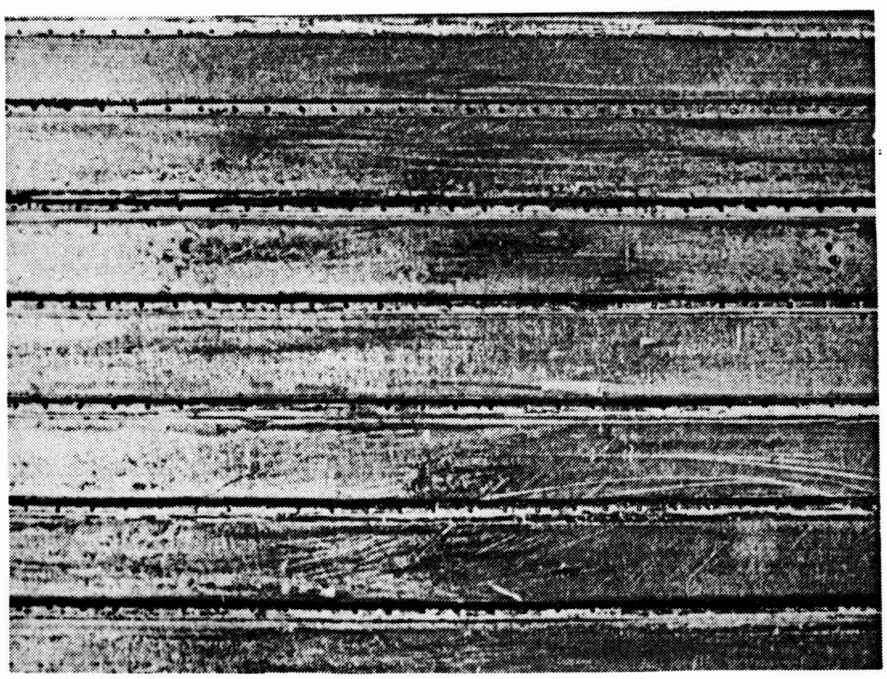

Figure 84. Close-up View of Clogged Metering Holes from Improper Cure Cycle During Verification Bond Run

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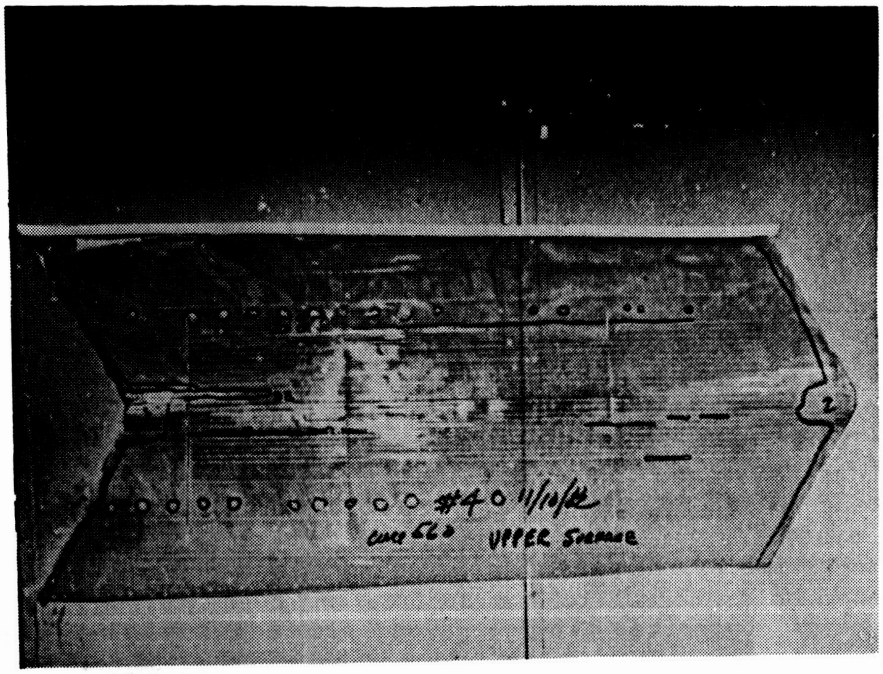

Figure 85. Fourth Verification Bond Run on Roll Formed Skin. Marks Denote Areas Where Two Layers of Adhesive were Required

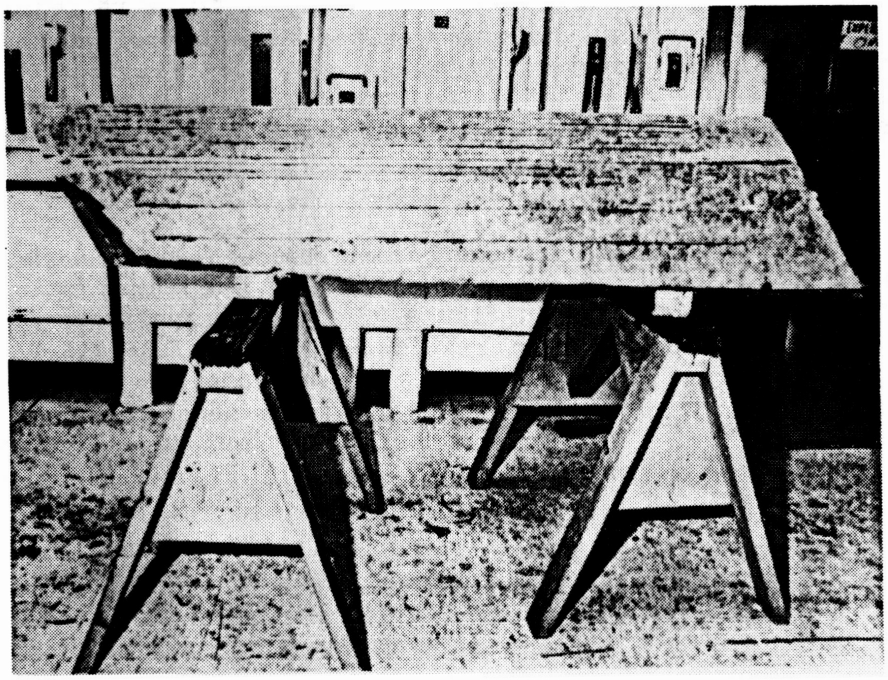

Figure 86. Flight Articles with Adhesive Applied Ready for Skin. Fuzzy Appearance is the Positioning Film Used to Control Adhesive Flow

After bonding, the skin was thoroughly inspected by coin tapping - no unbonded areas were found at that time. Figure 87 shows the article with the skin bonded in place.

A proof test was required prior to delivery. In order to load the unit, load pads had to be bonded to the skin. It was not considered prudent to try this operation after slotting so the proof test was accomplished as the next step. Figure 88 shows the article being proof tested.

4.2.4.2 Instrumentation

Static pressure taps were installed as detailed in the engineering drawing. The taps had to be installed in concert with the suction system since there was no space to accomplish the instrumentation task either before or after the suction system installation. Holes had been drilled through the substructure at the same time as the metering holes were drilled. Drilling 0.0881 cm (.032 in) diameter holes normal to the surface and over 1.27 cm (0.50 in) deep through a mixture of composite, fiberglass and glue could only be accomplished with automatic equipment. It was also planned to use the static pressure taps at the end of each slot as a line up point for slotting. Figure 89 shows the flight article with all the surface static pressure taps installed. Hot film gauges had also been installed by LaRc personnel at this point. It should be noted that while the tap positions were located and drilled on the exterior surface from a master mylar template developed from loft data, some misalignment occurred while back drilling the holes from the inside. During bonding the predrilled holes filled with adhesive and had to be reopened. In most cases the drill followed the old hole and exited through the titanium skin in the exact location desired. However, in some cases, the drill tended to lead off causing a slightly mislocated static pressure tap. The exact position of each tap was measured during the functional test and these data supplied to LaRC.

4.2.4.3 Suction System Installation

A hint of problems to come became apparent while plumbing the sonic fatigue article as shown in Figures 90 and 91. The first step was to install the suction nipples as shown in Figure 92. These nipples were bonded in with Hysol EA9209.1 adhesive and then reinforced with a fiberglass/epoxy collar. As a final step they were overcoated with adhesive to seal them. Following installation of the suction nipples, suction tubes were installed as shown in Figure 93. With all nipples, surface tubes and instruments installed, rubber hoses were connected to the nipples as shown in Figure 94. The next step was to install the forward diaphragm followed by the row of piccolo tubes and through tubes directly aft of the forward diaphragm. Figures 95 and 96 show these steps. A row of tubes was then located just forward of the sloped diaphragm followed by the diaphragm, and a final layer of tubes. Figures 97 and 98 show the completed assembly.

4.2.4.4 Slotting

The slotting operation is a dirty procedure with small chips being generated in a copious bath of cutting fluid. It was decided to delay slotting the flight test article until after the plumbing installation so as to have a means of continuously back flushing the slot as it was being cut, also, with the

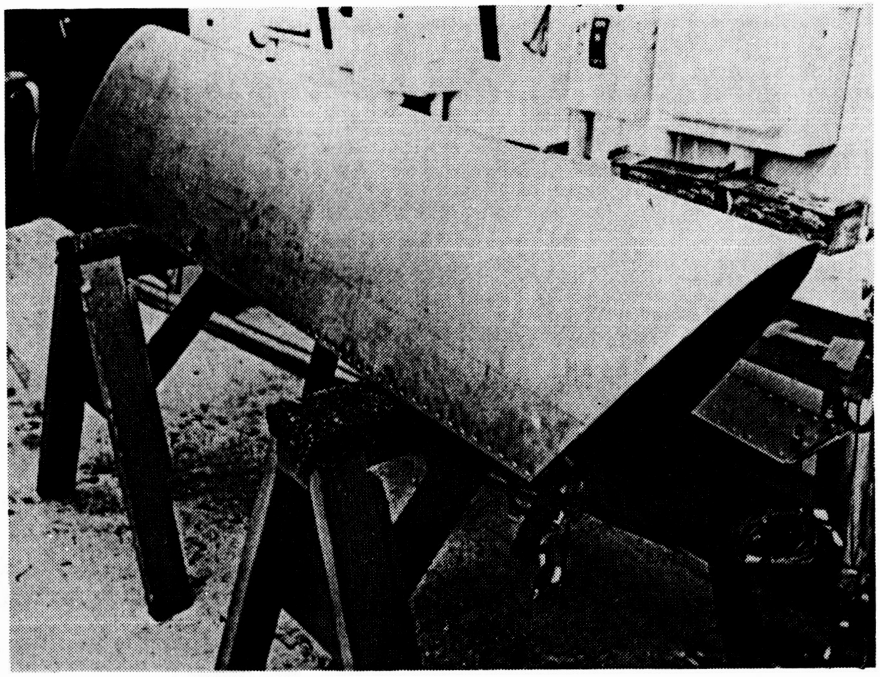

Figure 87. Flight Article After Bonding Skin to Substructure

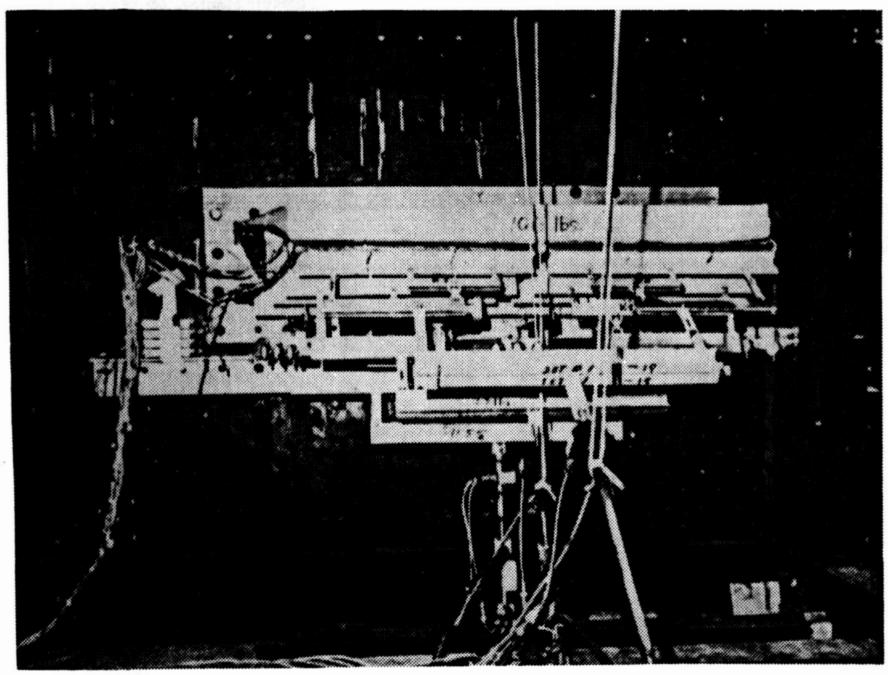

Figure 88. Flight Article Being Proof Tested

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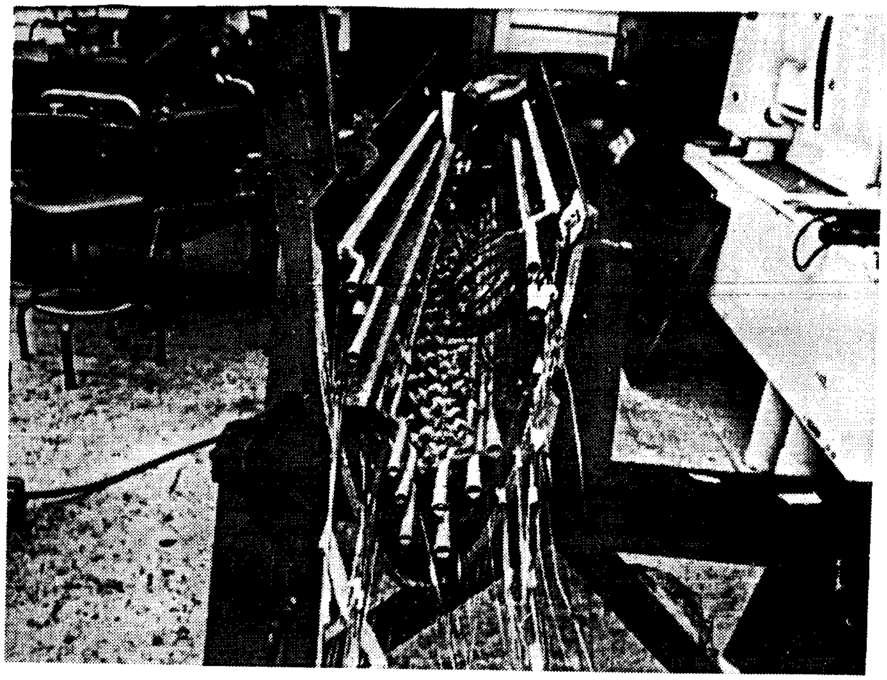

Figure 89. View of Flight Article Showing Static Pressure Lines Installed

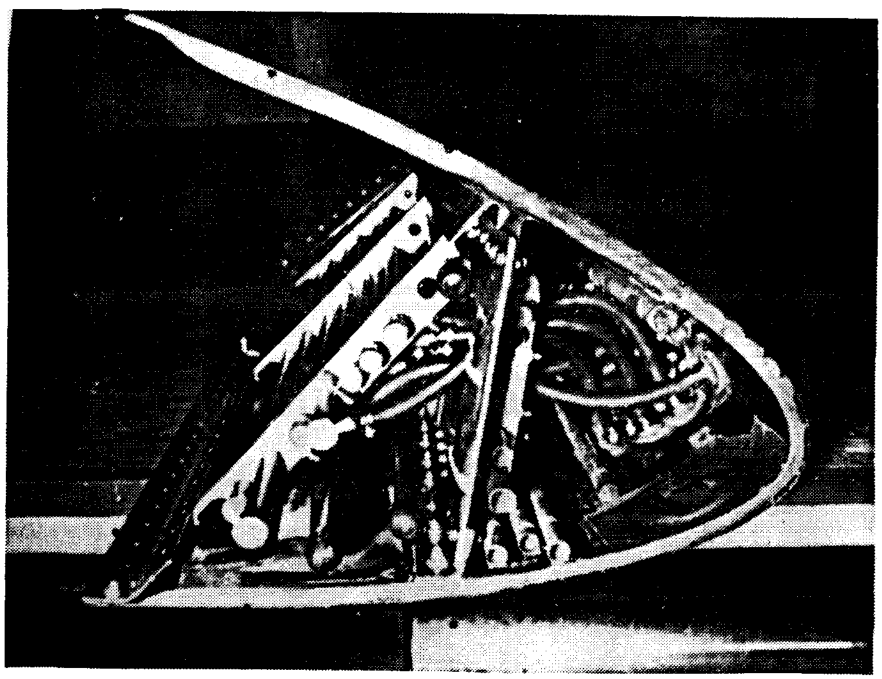

Figure 90. Sonic Fatigue Article Showing Inboard End with Plumbing Installed

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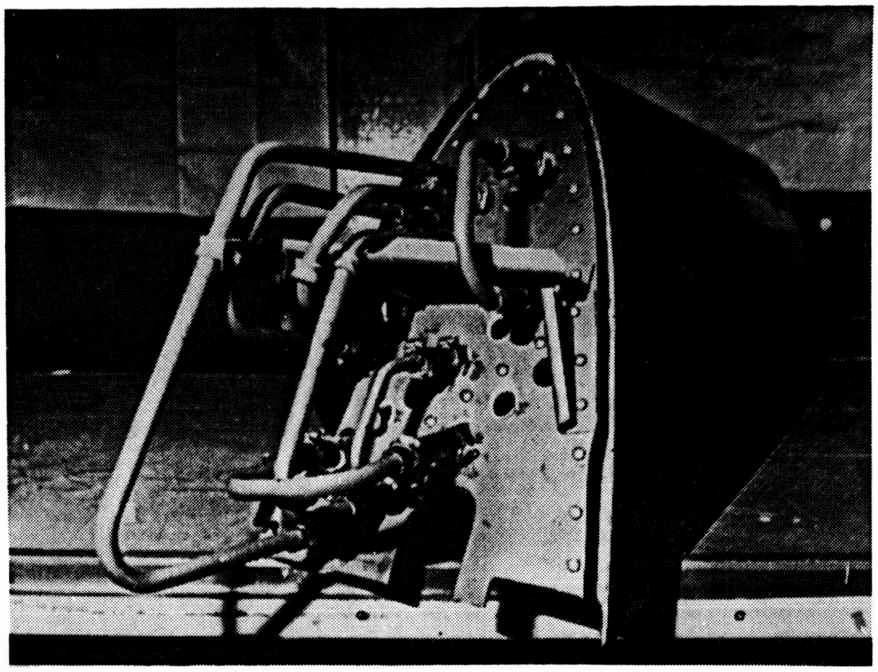

Figure 91. Outboard End of Sonic Fatigue Article

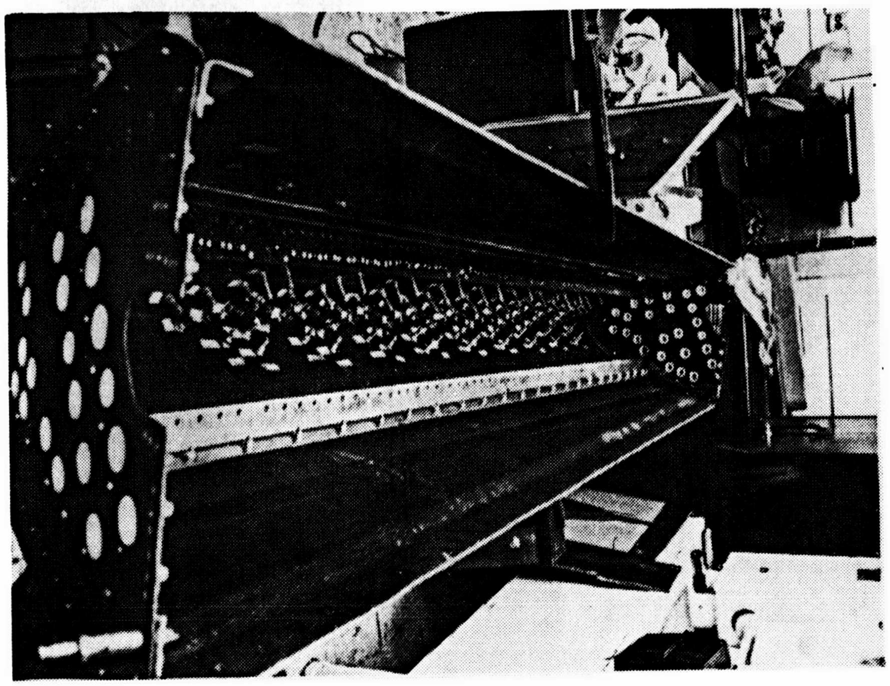

Figure 92. Flight Article with Suction Nipples Installed

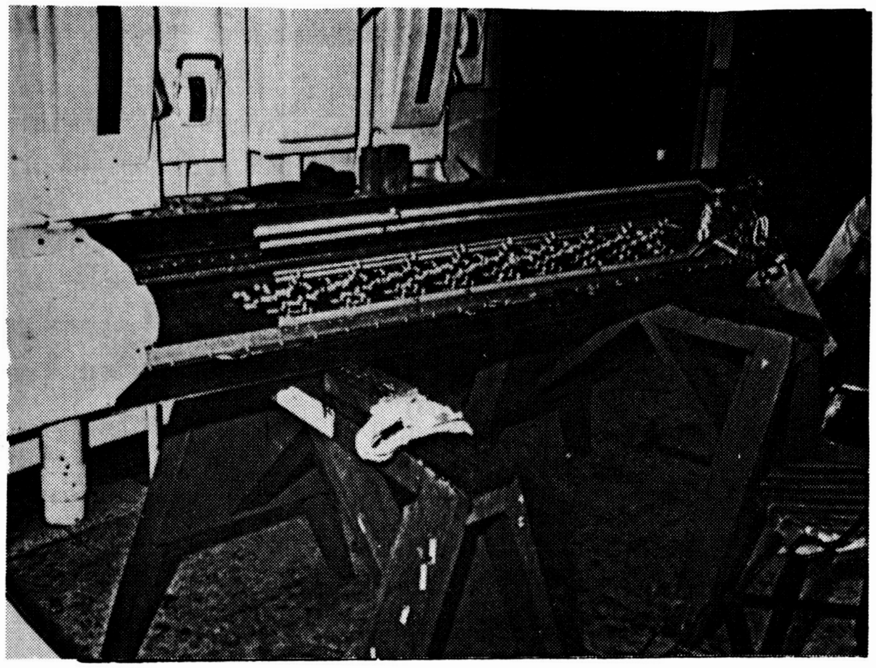

Figure 93. Flight Article with Part of Suction Tubes Installed

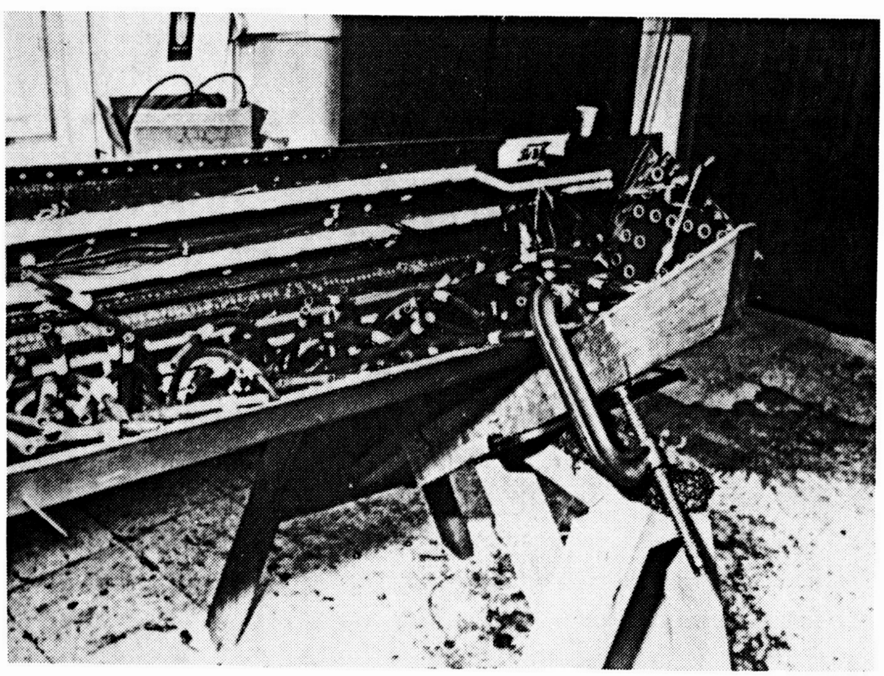

Figure 94. Flight Article with all Nipples, Surface Tubes, Instrumentation Installed with Hoses Connected to Nipples

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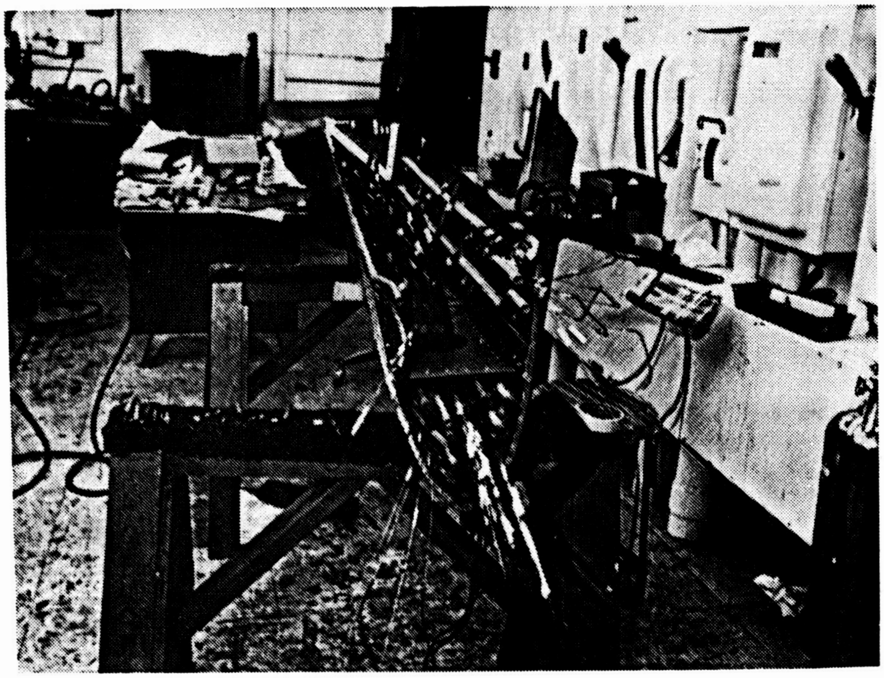

Figure 95. View Showing Forward Diaphragm Installed

Figure 96. View Showing Row of Piccolos Tubes and Through Tubes Forward of Aft Diaphragm

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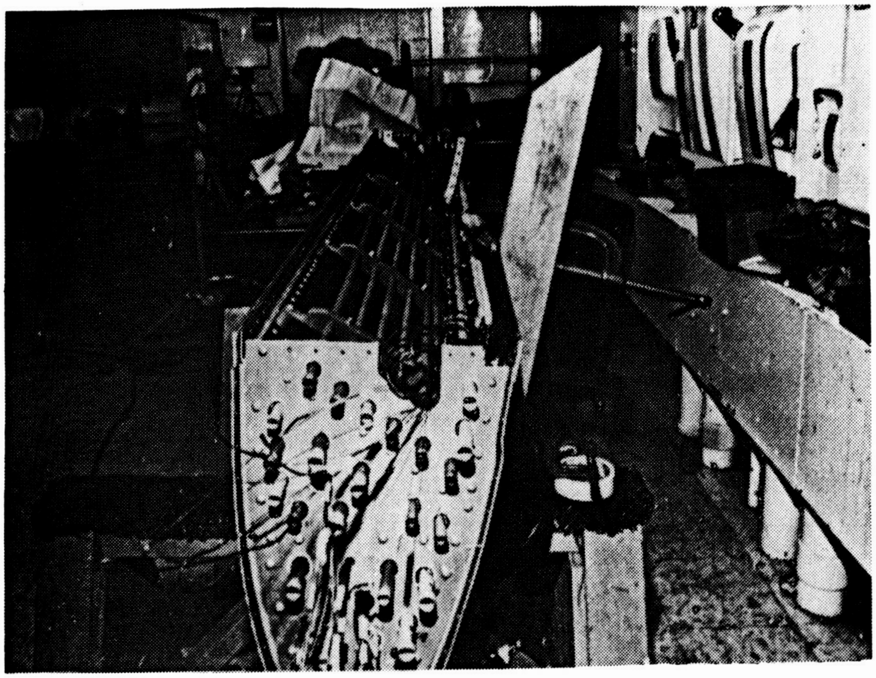

Figure 97. View of Inboard End of Flight Test Article

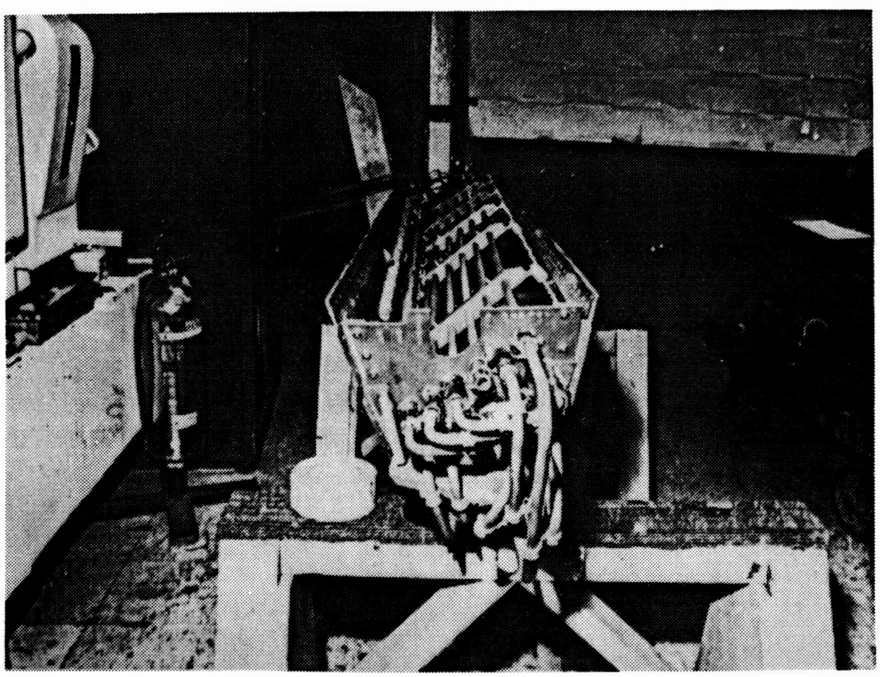

Figure 98. View of Outboard End of Flight Article with all Plumbing Except One Loop Tube Installed

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diaphragms in place and the structure rigid, there would be less tendency for the structure to move or the skin to springback thus changing slot width. A 39.5 KPa (5 psi) regulated air source was connected to each slot as it was cut. This procedure seemed to work well in keeping the slot clean. Slots were cut exactly as developed during the MD2 test, with a saw speed of 64 RPM and a feed of approximately 10.16 cm (4 in) per minute being used. The tool try article and reslotting of the sonic article was carried out on an old Lucas Boring mill. While this seemed to do an adequate job some saw breakage did occur. This saw breakage was determined to be primarily caused by rough feeding due to a worn feed system. A new computer controlled mill that was placed in operation in the shop prior to slotting the flight test article was used. This mill produced significantly better results. Alignment of the saw over the center of the slot was accomplished by locating to the static pressure tap at each slot end. As a result of the leading edge taper the saw could not cut normal to the surface from one end to the other. It was therefore lined up normal at one end and forward at the other end. Figure 99 shows a mechanic and machinist making this rather delicate alignment. Figure 100 shows a slot being cut on the flight test article. As each slot was cut it was tested for air flow and then sealed with tape to prevent contamination. A close up view of the jewelers saw is shown in Figure 101. This view also shows the slot lineup scribe markers. Note also that the static pressure taps are protected by tape.

Saw breakage was not a problem during slotting of the flight test article. A saw was broken but was caused by a computer malfunction on the depth feed. The cutting of precision slots in titanium was developed to a relatively high technology during the program and was one of the major accomplishments of the program.

4.2.5 Repairs/Rework

During airflow tests on the first slot cut it became apparent that there was flow blockage. In investigating the cause, part of the skin was removed from the quality control test panel which had been bonded at the same time using the same adhesive as the flight test article. This showed that some adhesive flow had occurred into most of the slot ducts. In areas where the slots were farther apart more adhesive was available and had blocked the slot duct completely. A section was cut from the panel with the skin intact and polished. Figure 102 is a photomacrograph of this section. Note the extensive fillet that extends almost across the titanium which was wet by the adhesive readily since it was primed. It was later found that this film of adhesive, which was cut by the saw almost as if by a knife, would reseal the slot to air flow as if it were a rubber flapper seal. Figure 103 shows a view of the surface of the panel in an area with 1.57 cm (0.62 in) slot spacing. Note that the duct has a heavy fillet but is not blocked. In areas where the slot spacing was 2.54 cm (1.0 in) or more there was adequate adhesive to completely block the duct. The investigation also revealed that the new lot of adhesive was "greener" than normal and had excessive flow for the system. Such flow had never been experienced by Lockheed-Georgia in over 15 years of using the FM123-4 adhesive system. It could have been prevented by requiring a flow limit in the specification but past experience did not indicate the need for such a requirement.

Several approaches were considered and investigated for correcting the flow blockage short of a complete reskinning of the flight test article. Blockage in the forward, closely spaced slots was less than that for the aft slots. The

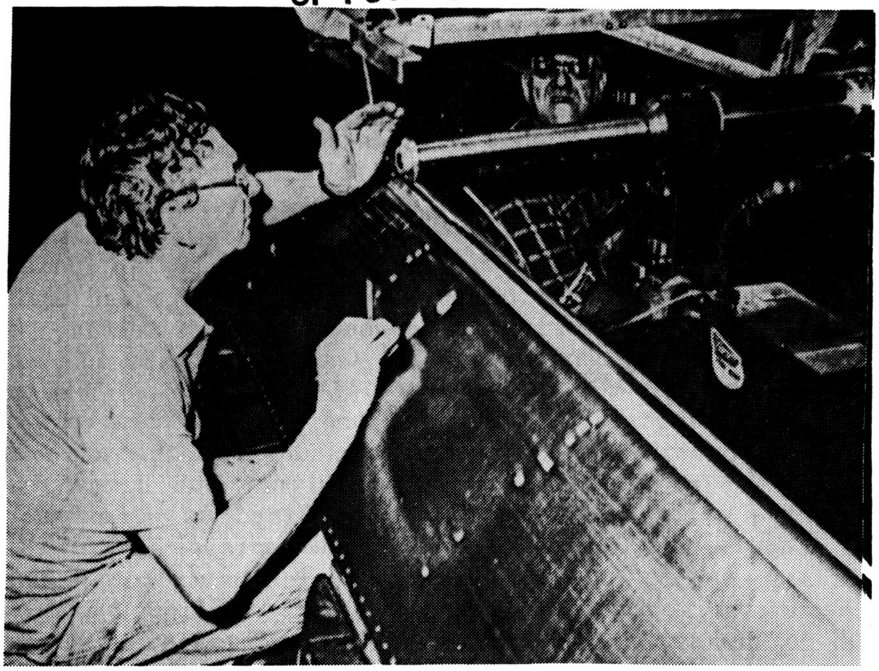

Figure 99. View Showing Saw Being Aligned with Slot Line-Up Markers

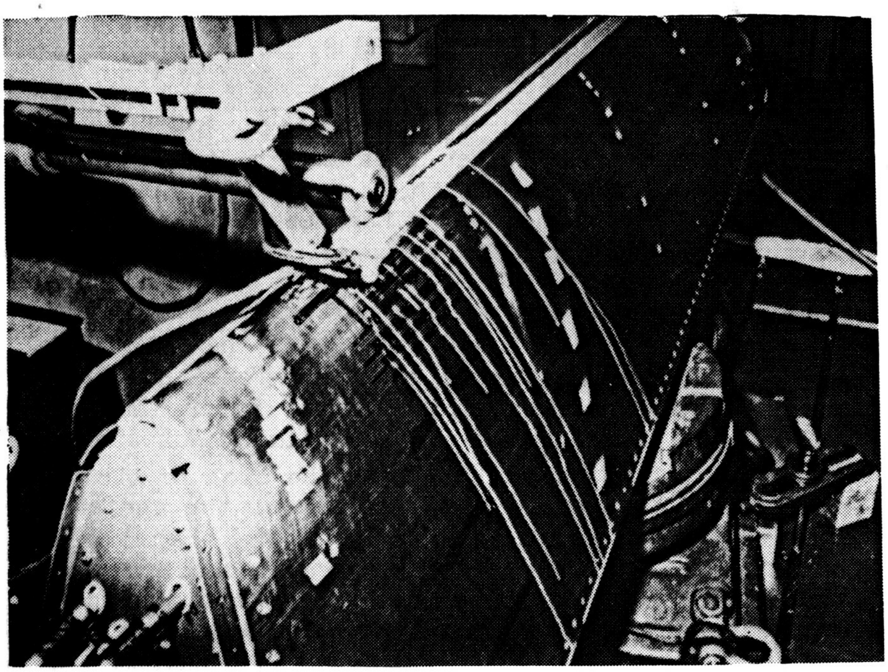

Figure 100. Slot Being Cut with 0.0076 cm (0.003 In) Thick Jewelers Saw

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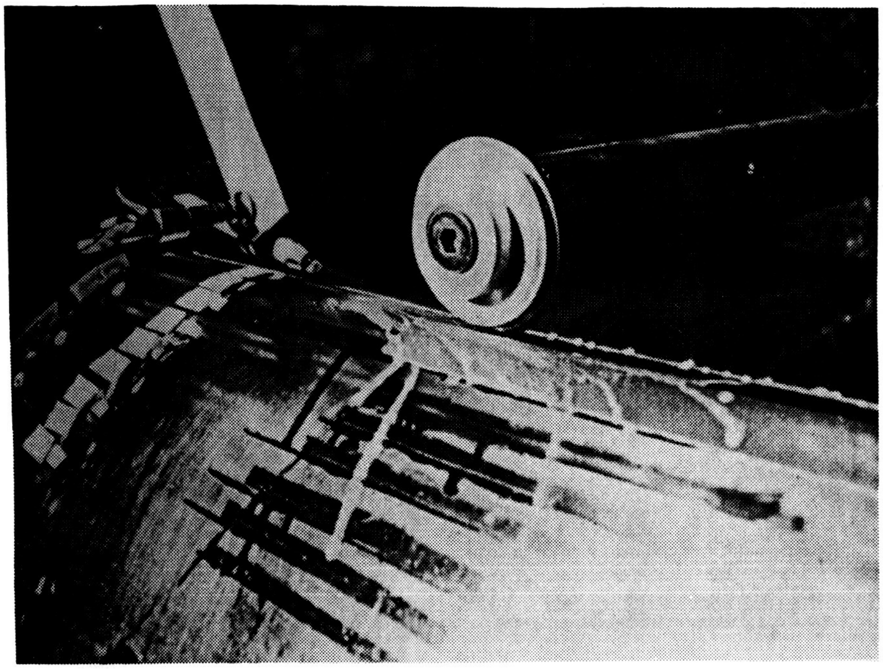

Figure 101. Close-Up View of Slot Cutting

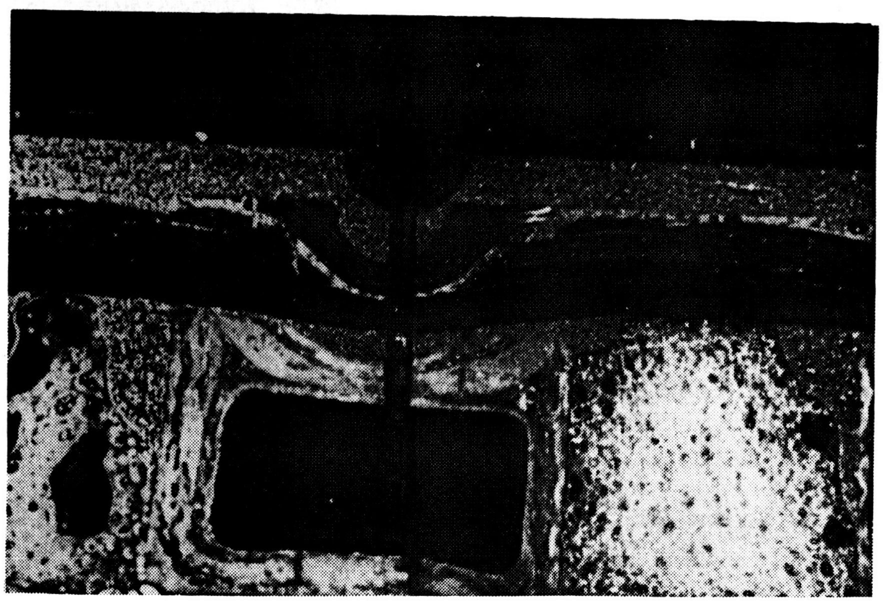

Figure 102. Photomacrograph of Section Through Slot Duct and Collector Duct Showing Adhesive Flow

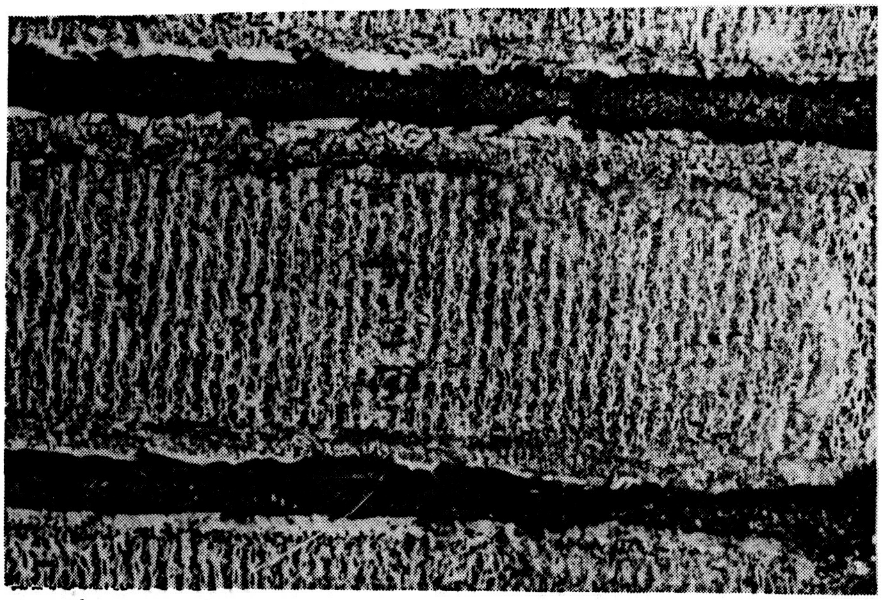

Figure 103. Photomacrograph of Section of Test Panel with Face Sheet Removed

first approach was to cut a groove through the skin and open the end of the slot duct. This was easily accomplished using the same setup as was used to cut the slot ducts. Figure 104 shows the access grooves into the slot ducts. A 0.127 cm (0.050 in) diameter hypodermic tube was chucked into a drill and used to ream out the slot duct, Figure 105. In most cases, the hypodermic tubing could be freely run from one end of the slot duct to the other, which verified that most of the blockage was caused by the adhesive fillet on the skin which had to be gradually worn away by the drill. This operation continued until acceptable flow was achieved. Slot D3 could not be recovered as it had been cut out of the slot duct by a machine malfunction.

Slots U8, U9, U10, U11, L6, L7 and L8 were completely blocked and could not be repaired in the manner used on the other slots. A possible solution was to remove a strip over the slot, clean the duct and replace the strip. This procedure was tested using the sonic fatigue article. Figure 106 shows strips removed by cutting with the same jewelers saw used for slotting and then peeling the titanium from the substructure. A replacement strip was rolled to contour, hand fitted to the gap, and bonded under vacuum pressure. Heat was applied with flexible strip heaters as shown in Figure 107. After bonding, the area was filed smooth and reslotted. Several techniques were developed which allowed a smooth surface to be obtained. Prior to making the repairs, a procedure to reduce adhesive flow was developed. It was found that by aging the adhesive at 37.8°C (100°F) for one hour the flow could be reduced to the level previously experienced. Several strips were bonded on the sonic fatigue test article and removed to verify that no blockage would occur during the repair operation.

After a demonstration to NASA personnel, it was agreed to use this procedure to rework all completely blocked slots except slot U8. Slot U8 did not

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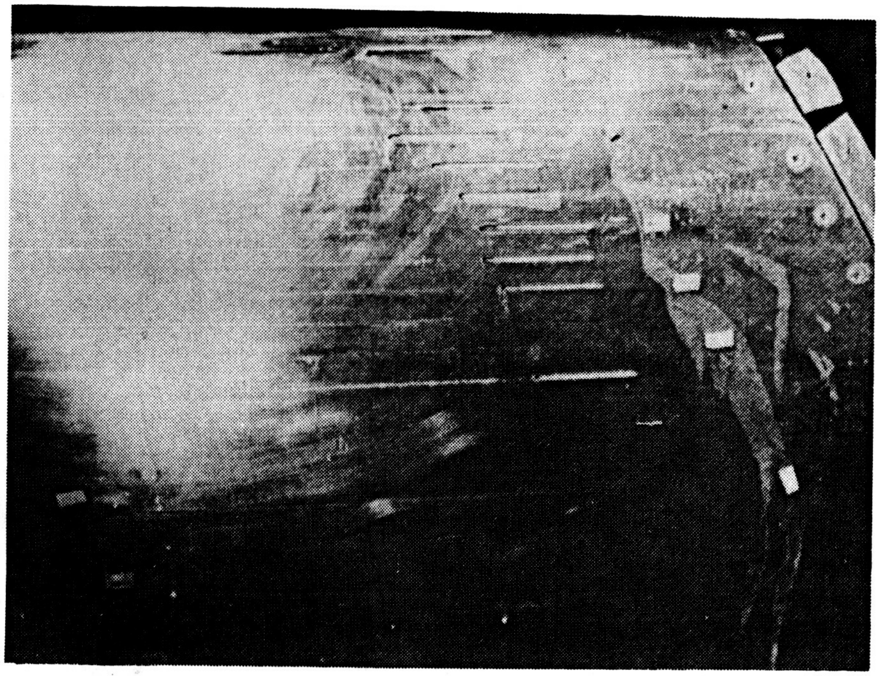

Figure 104. Photograph of Flight Article with Access Grooves to Slot Ducts

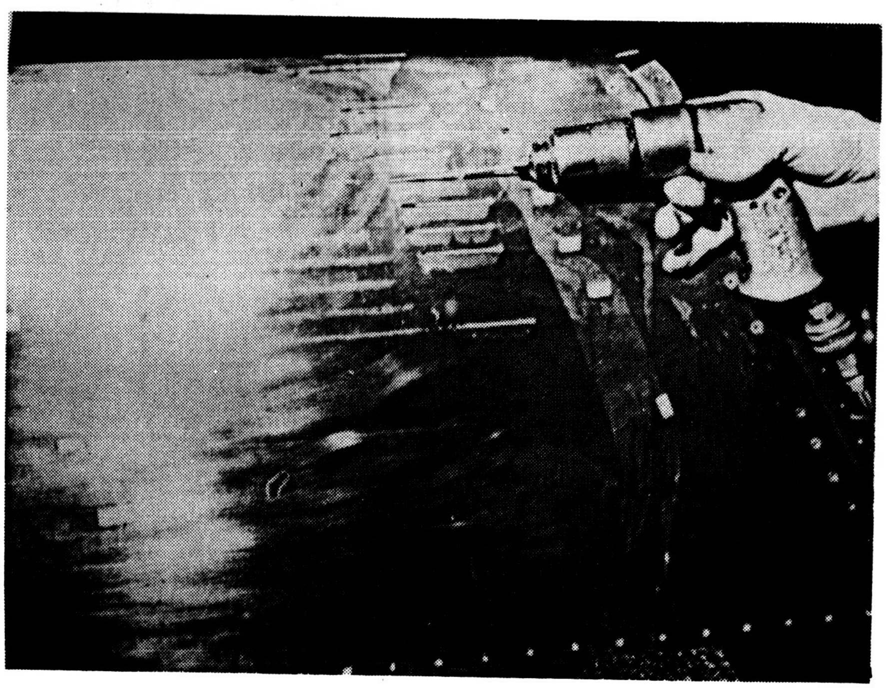

Figure 105. View Showing Excess Adhesive Being Removed from Slot Duct

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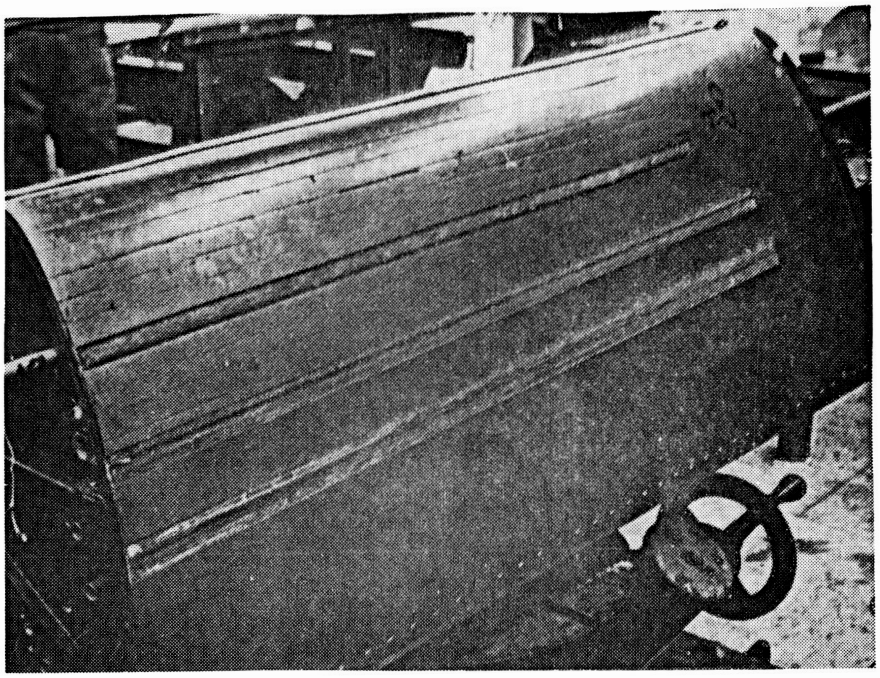

Figure 106. Strip Replacement Test on Sonic Fatigue Test Article

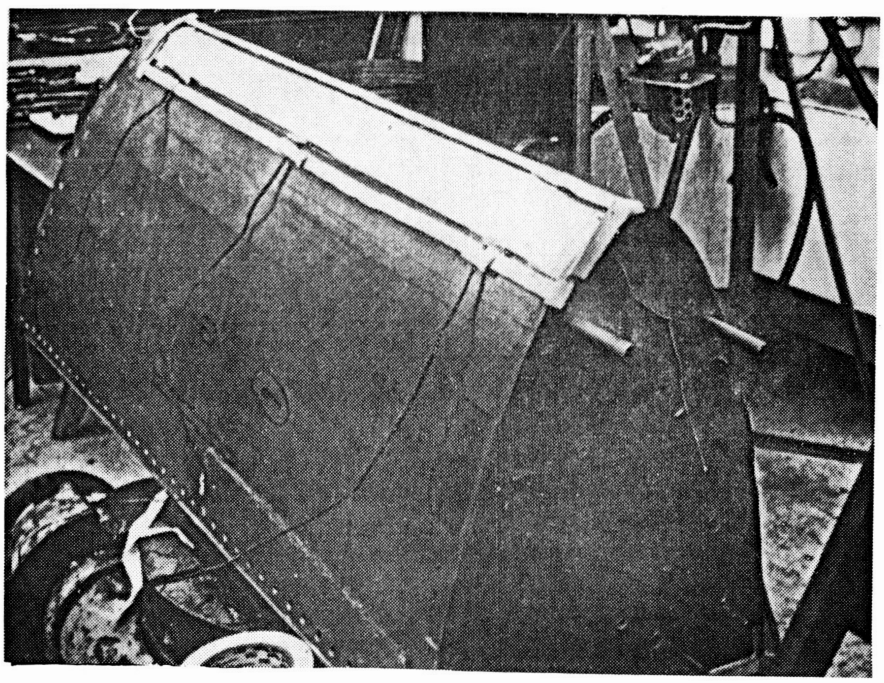

Figure 107. Repair Test Development Showing Local Strip Heaters for Bonding Strips in Skin

have adequate bond area between slots U8 and U7 to effect a repair. Figure 108 shows the flight test article being reworked. Figure 109 shows the almost total blockage in slots U9 and U10. After the titanium strips were removed, the slot duct was recut to remove adhesive. Metering holes were cleared by hand drilling as required. The removal clearing, rebonding, and slotting operation then proceeded very smoothly.

4.2.6 Functional Testing

Following all repairs a complete functional test of the suction system was conducted. Slot widths were measured using the Wilson air gauge as shown in Figure 110. During this operation it was discovered that the Wilson air gauge was an excellent means of measuring blockage of the slot caused by an adhesive fillet. A 0.0076 cm (0.003 in) feeler gauge would past freely through all slots cut with a 0.0076 cm (0.003 in) saw, where a 0.0102 cm (0.004 in) gauge had to be forced into the slot. However, since the Wilson air gauge is measuring the effective width to air flow it shows this type of blockage. Results of slot width measurements are shown in Table 2. In addition, the location of each slot in relation to the surface loft mylar master was determined and is shown in Table 3.

Surface smoothness was measured using the Lockheed Ripple Measuring Unit as shown in Figures 111 and 112. In general, the flight-test article was within the wave tolerance criteria.

After all repairs were complete a delaminated area was found on the upper surface as shown in Figure 113. This delamination was marked and a mylar overlay made for use at Dryden to track any growth. Likewise, a small delaminated area was found on the lower surface as shown in Figure 114. It was treated as the one on the upper surface. No delaminations were found during the initial coin tap inspection prior to the repair work. In all likelihood, the two

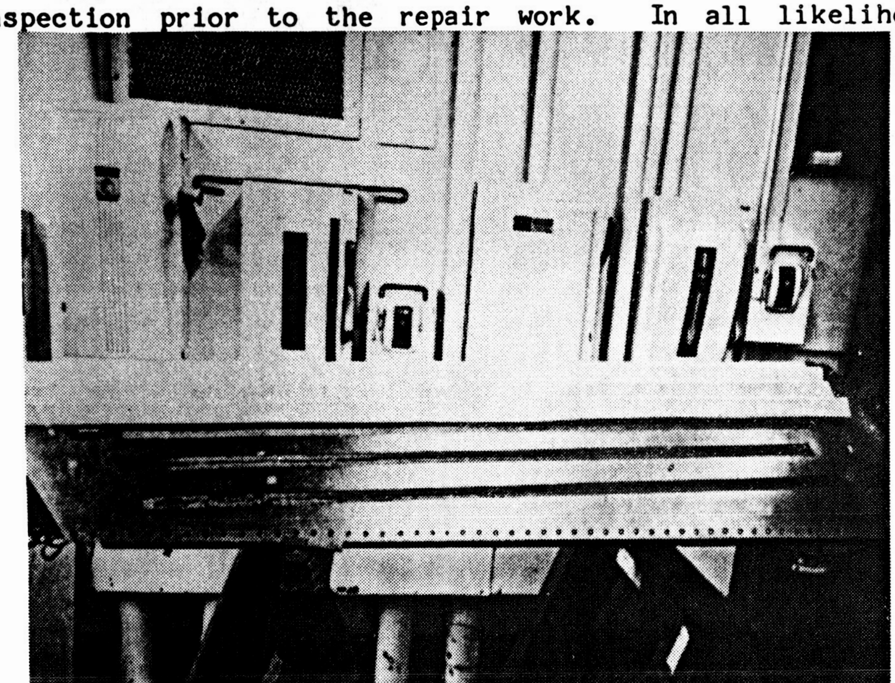

Figure 108. Flight Article with Strips Removed to Open Slot Ducts for Adhesive Removal

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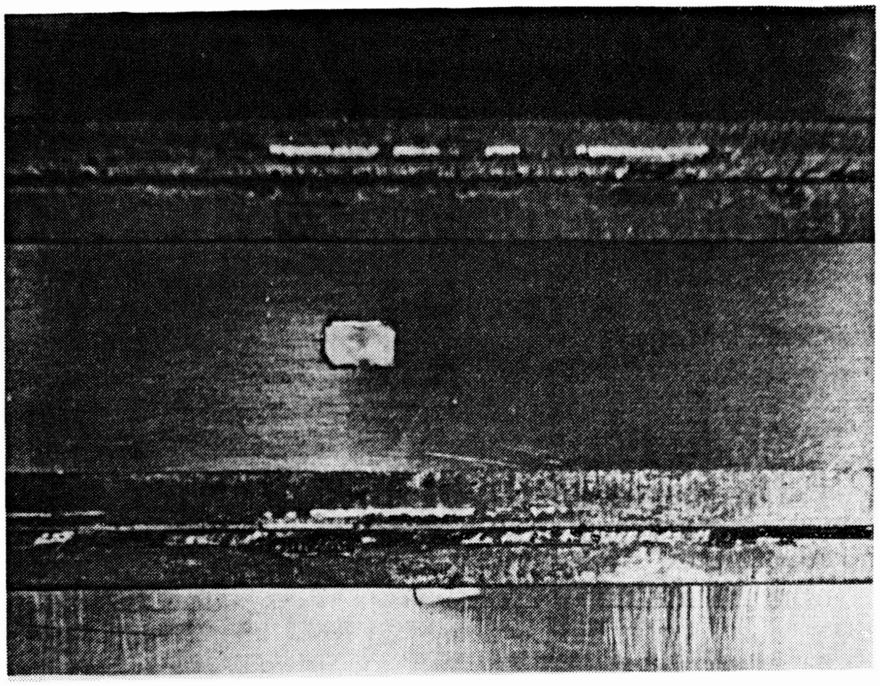

Figure 109. Close-Up View Showing Adhesive Blocked Slot Ducts

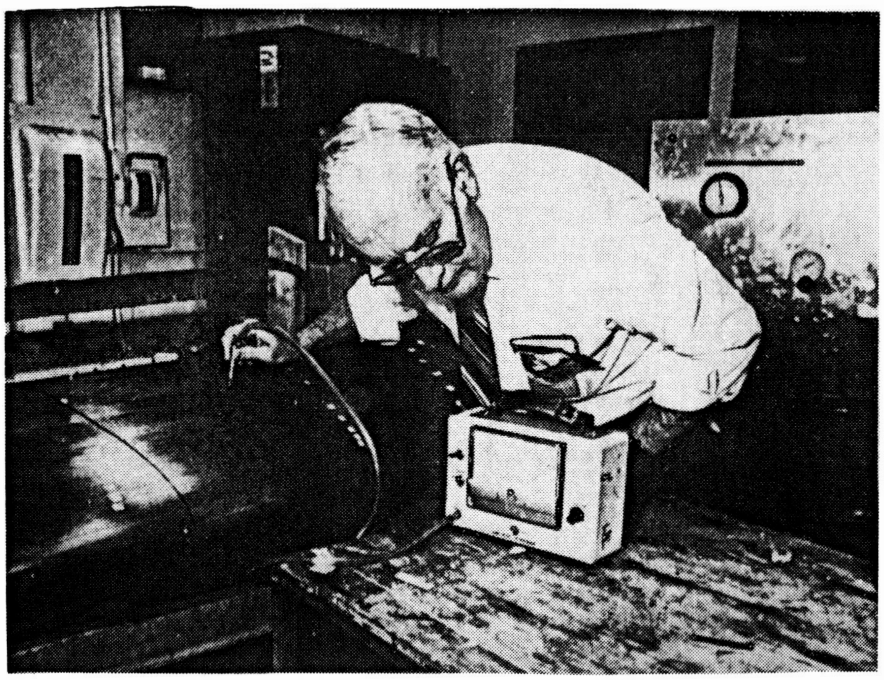

Figure 110. Slot Width Measurement with Wilson Air Gauge

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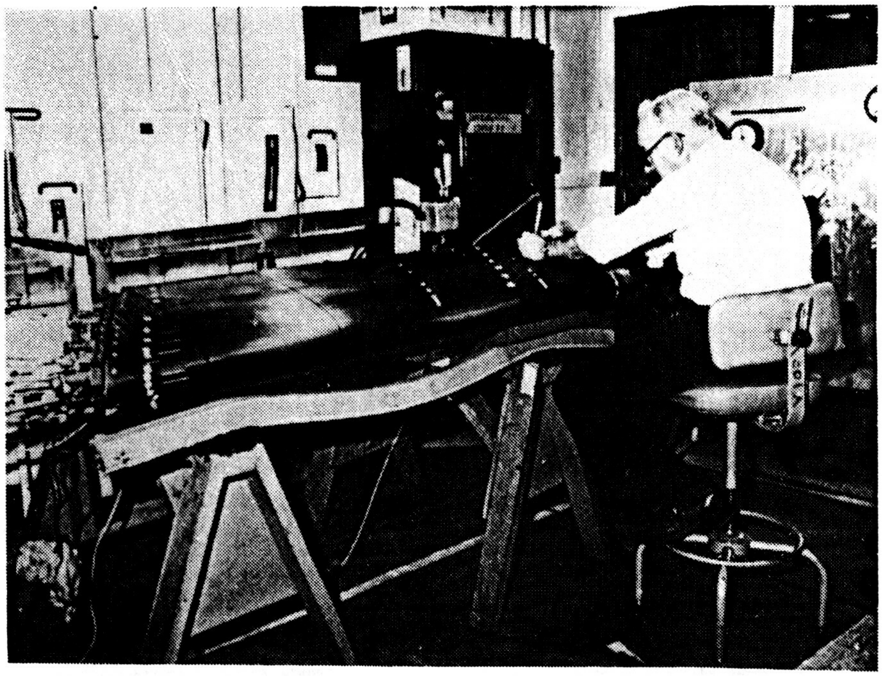

Figure 111. Measuring Smoothness with Ripple Measuring Unit

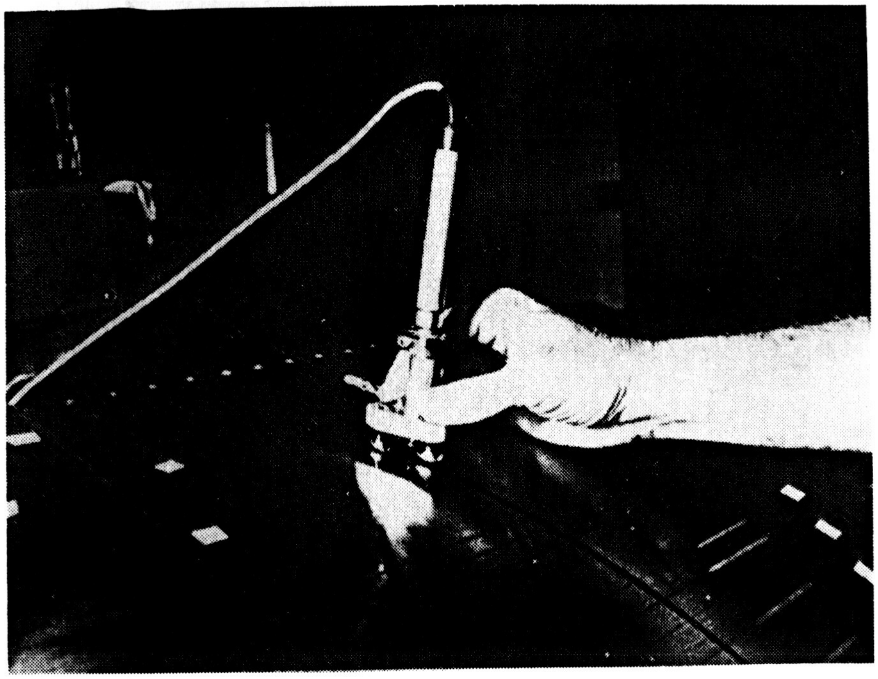

Figure 112. Close-Up View of RMU Probe in Use

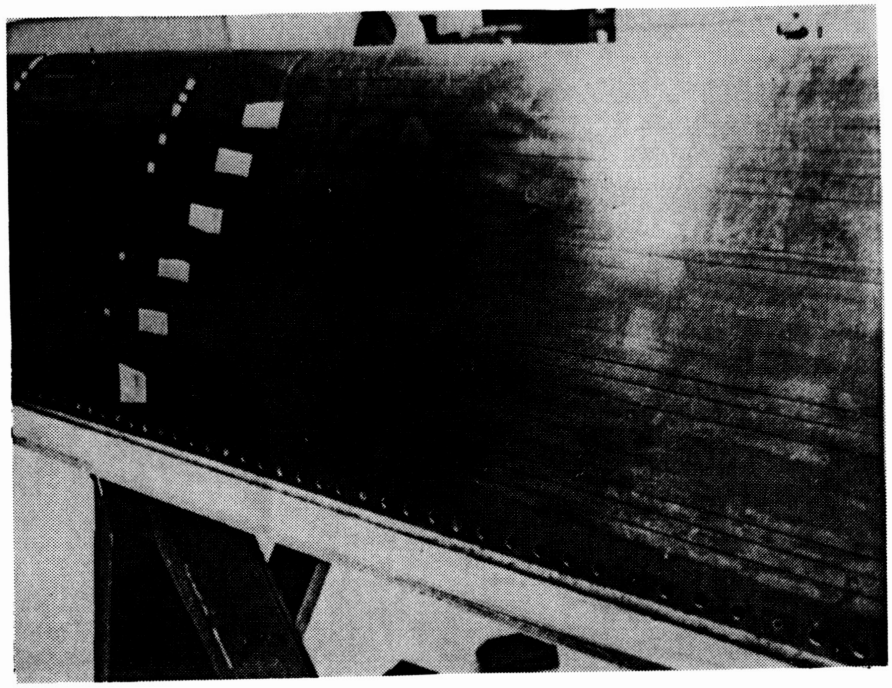

Figure 113. View of Upper Surface with Delaminated Area Identified

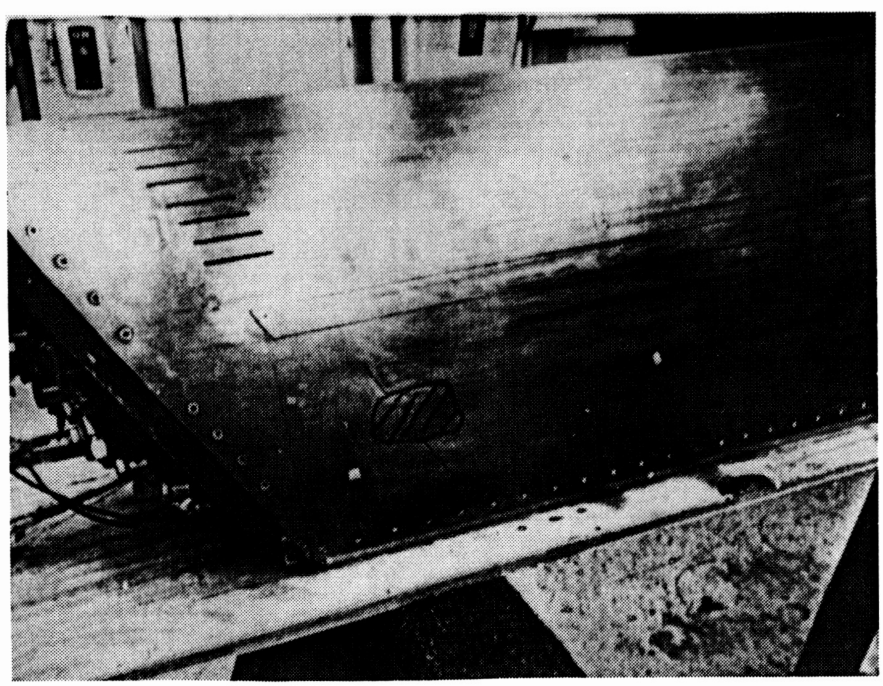

Figure 114. View of Lower Surface Showing Skin Delamination

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TABLE 2. SLOT WIDTH MEASUREMENTS

DIST FROM INBOARD END OF SLOT INCHES	SLOT NUMBER SLOT WIDTH IN INCHES x 10^{-3} C1	C2	D1	D2	D4	D5	D6
2	2.2	3.0	3.2	4.1	4.4	3.2	3.5
4	2.4	2.9	3.2	3.1	4.1	3.5	3.2
6	2.4	2.9	2.4	3.2	3.6	3.1	3.2
8	2.5	3.0	3.2	3.6	4.0	3.2	3.6
10	2.4	2.7	3.3	3.3	4.8	2.6	3.5
12	2.1	2.8	3.5	3.2	3.8	3.1	3.5
14	2.6	2.6	2.6	3.4	3.4	3.2	3.3
16	1.9	2.6	3.2	3.4	3.0	3.3	3.4
18	2.1	3.1	2.8	2.9	3.3	3.5	3.6
20	3.2	2.0	2.6	3.3	3.1	3.8	3.2
22	2.3	2.4	2.7	3.0	3.0	3.5	2.6
24	2.3	2.2	2.7	3.1	2.9	3.1	2.3
26	2.2	2.3	2.3	3.2	2.9	3.0	2.5
28	2.2	2.3	2.3	3.3	2.9	2.9	3.1
30	2.3	2.2	1.8	3.2	2.6	3.0	3.2
32	2.0	1.8	1.8	3.1	3.0	3.1	3.0
34	2.0	2.0	2.1	3.1	2.6	3.0	3.1
36	2.0	2.1	2.0	2.9	2.9	2.7	3.1
38	2.4	2.3	2.4	3.3	3.1	3.1	3.2
40	2.4	2.2	2.5	3.3	3.8	3.4	3.1
42	2.1	1.5	3.0	2.8	2.6	3.4	3.5
44	2.1	2.0	1.9	3.1	2.8	3.6	3.5
48	3.5	2.2	1.8	3.2	2.2	3.7	3.5
50	2.1	1.7	2.2	3.3	2.2	3.6	3.5
52	2.1	1.8	2.0	3.4	2.4	3.6	3.5

DIST FROM INBOARD END OF SLOT INCHES	SLOT NUMBER SLOT WIDTH IN INCHES x 10^{-3} U1	U2	U3	U4	U5	U6	U7	U9	U10	U11
2	3.2	3.1	2.5	3.5	2.9	2.4	3.4	4.8	4.9	4.3
4	3.4	3.2	2.7	3.8	3.0	4.1	3.6	4.9	4.6	4.1
6	2.7	3.4	2.8	3.5	2.4	4.2	3.4	4.9	5.1	4.1
8	3.1	2.5	3.0	3.6	2.4	4.0	3.4	4.8	5.0	4.5
10	3.0	2.4	2.9	3.4	2.4	3.8	3.3	5.0	4.0	4.0
12	3.4	3.1	3.0	3.4	2.6	3.8	3.0	5.0	5.1	4.0
14	2.9	2.7	3.1	3.8	2.3	3.9	3.3	4.4	5.1	4.1
16	2.7	3.2	3.2	3.6	2.3	3.3	3.3	4.6	4.6	4.1
18	2.8	3.1	3.1	3.5	2.1	3.8	3.2	4.8	4.6	3.6
20	3.7	3.3	2.0	3.1	2.1	3.4	2.8	4.8	4.0	3.6
22	2.9	3.1	2.4	3.6	2.3	3.8	3.1	4.7	4.2	3.3
24	2.9	3.0	2.9	3.4	2.4	3.9	3.1	4.7	4.1	3.2
26	2.5	3.0	2.9	3.3	3.0	3.5	3.0	4.7	4.1	3.3

TABLE 2. SLOT WIDTH MEASUREMENTS (CONT'D)

DIST FROM INBOARD END OF SLOT INCHES	SLOT NUMBER SLOT WIDTH IN INCHES x 10^{-3}									
	U1	U2	U3	U4	U5	U6	U7	U9	U10	U11
28	3.0	3.2	3.0	3.5	2.4	3.4	3.1	4.9	4.4	4.4
30	2.5	2.9	2.7	3.4	2.2	3.2	2.9	4.7	3.9	3.7
32	2.3	2.7	3.2	3.3	3.0	3.4	3.6	4.7	3.8	3.8
34	3.4	3.0	3.2	3.3	2.5	3.5	3.3	4.0	4.3	3.6
36	3.2	2.5	3.2	3.6	2.6	3.6	3.3	4.6	3.8	3.8
38	3.1	2.9	3.2	3.8	2.4	2.9	3.6	4.2	4.4	4.3
40	3.0	3.1	3.3	3.4	2.3	3.5	2.6	4.4	4.6	4.2
42	2.8	3.4	3.0	3.3	2.3	3.7	2.9	4.7	4.2	4.4
44	2.8	3.3	3.1	3.5	2.6	3.7	3.0	4.6	4.2	3.9
46	2.9	3.2	3.1	3.2	2.3	3.3	3.3	4.7	4.6	3.9
48	2.6	3.1	3.1	3.3	2.3	3.2	2.8	4.5	4.5	4.2
50	2.9	3.0	3.2	3.6	2.5	3.1	3.2	9.0	4.8	4.2
52	2.9	2.9	3.3	3.2	2.6	3.2	3.4	3.2	4.5	4.4

DIST FROM INBOARD END OF SLOT INCHES	SLOT NUMBER SLOT WIDTH IN INCHES x 10^{-3}							
	L1	L2	L3	L4	L5	L6	L7	L8
2	3.6	1.9	3.8	3.1	3.9	3.6	4.0	5.1
4	3.9	1.6	3.9	3.6	4.1	3.4	4.0	4.8
6	3.7	1.7	3.9	4.0	3.1	3.6	4.2	4.0
8	3.8	3.1	4.0	3.8	3.2	3.9	4.1	4.2
10	3.3	2.2	3.6	3.5	3.3	3.8	4.2	5.0
12	3.4	1.8	4.1	4.0	3.5	3.5	3.9	4.1
14	3.7	1.6	4.0	3.1	4.0	3.1	4.0	4.8
16	3.4	2.0	3.7	3.8	4.0	3.0	3.8	4.8
18	3.1	3.0	3.9	4.0	3.9	3.5	4.0	5.0
20	3.0	3.5	4.0	3.8	3.9	3.4	3.5	4.6
22	2.2	2.7	4.0	4.0	4.0	3.3	4.0	4.7
24	2.8	3.0	3.9	3.9	4.5	3.3	3.4	4.8
26	2.9	2.7	4.1	3.8	4.2	3.3	3.7	4.8
28	2.8	3.8	4.1	3.7	4.4	3.1	3.9	4.8
30	2.8	3.8	3.7	3.4	3.3	3.5	3.9	4.7
32	1.5	3.5	3.7	3.9	3.7	3.4	3.6	4.5
34	2.0	3.6	3.9	3.7	3.8	3.5	3.6	4.3
36	1.0	3.6	3.4	3.4	4.4	3.5	3.7	4.4
38	1.0	3.8	3.3	3.0	4.3	3.5	3.7	5.1
40	1.0	3.2	3.2	3.1	4.4	3.3	3.9	4.2
42	1.0	3.6	3.2	4.0	4.1	3.3	3.8	5.1
44	1.0	3.8	3.1	3.6	4.0	3.4	3.5	4.9
46	1.0	3.8	3.1	3.2	4.0	3.4	3.7	4.6
48	1.0	3.9	3.1	3.2	3.8	3.4	4.0	4.4
50	1.0	4.1	3.1	3.7	3.2	3.2	3.8	4.6
52	1.0	4.3	3.0	3.3	3.1	3.0	3.9	4.4

TABLE 3. LEADING EDGE SLOT LOCATIONS
(INCHES)

SLOT NUMBER	INBD.	OUTBD.
U11	.00	.00
U10	.03 AFT	.03 AFT
U9	.03 AFT	.00
U8	.00	.00
U7	.05 AFT	.03 AFT
U6	.05 FWD	.08 AFT
U5	.06 FWD	.00
U4	.00	.03 AFT
U3	.03 AFT	.02 AFT
U2	.03 FWD	.03 FWD
U1	.00	.03 AFT
D1	.03 FWD	.02 FWD
C1	.00	.00
C2	.00	.04 AFT
D2	.04 FWD	.00
D3	.02 AFT	.03 AFT
D4	.03 FWD	.05 AFT
D5	.07 FWD	.00
D6	00	.00
L1	.06 FWD	.00
L2	.06 FWD	.03 AFT
L3	.00	.02 FWD
L4	.00	.00
L5	.00	.00
L6	.03 AFT	.00
L7	.03 AFT	.05 FWD
L8	.04 FWD	.06 FWD

SLOT LOCATIONS IN COMPARISON TO SURFACE LOFT.

delaminations were caused by marginal bonds that unbonded during the rework handling. The strip of titanium removed for the U11 repair was on the edge of a delaminated area and the strip appeared to be contaminated at that spot. Adjacent areas did not have the contaminated appearance. No delaminations were present in the repaired areas and based upon the appearance of the titanium removed for the U11 repair, the delaminations probably will not grow.

Although slots U8 and D3 were lost, stability calculations performed by Lockheed and reviewed by NASA showed that laminar flow could still be achieved without these slots. Some blockage remained in other slot ducts and flow tests showed areas of low flow across the spanwise slots. Lockheed analysis shows, however, that the suction system is adequate to overcome these blockages and reduced flows to achieve laminar flow.

Figures 115 and 116 show the flight-test article being crated for shipment to NASA Dryden on April 25, 1983.

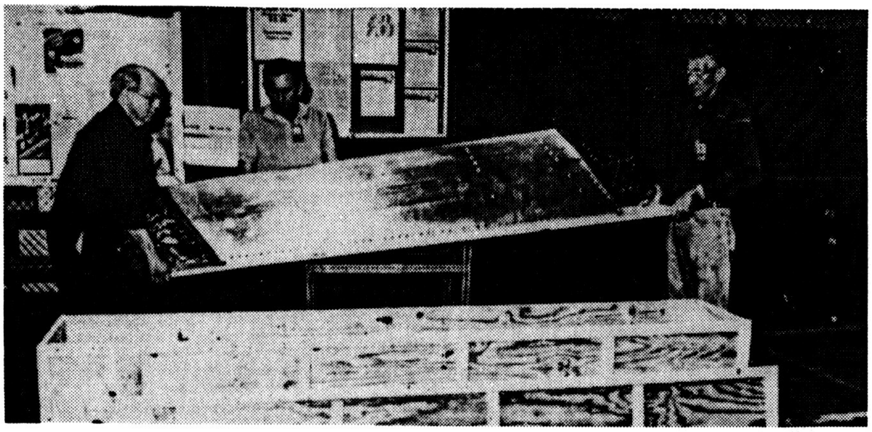

Figure 115. Flight Article Being Crated for Shipment

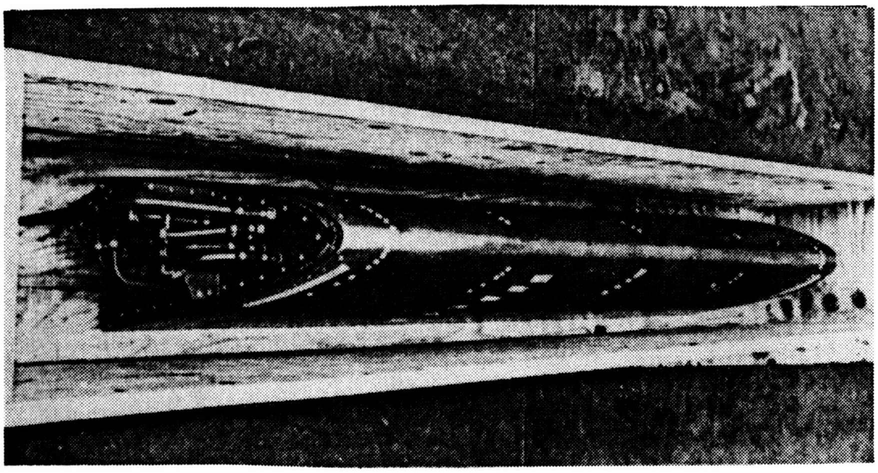

Figure 116. Flight Article in Crate Ready for Shipment

5.0 LFC SYSTEMS

The LFC systems include the Lockheed suction system, the McDonnell Douglas suction system, the turbocompressor suction system, the cleaning/anti-icing systems, the purge systems and the nitrogen pressurization system. These systems are required to support laminarization of the leading-edge test article. The Lockheed and McDonnell Douglas systems include the appropriate leading edge test article (LETA), chamber valves, the interconnecting lines between the LETA and the chamber valves, and the lines between the chamber valves and turbocompressor suction pump.

The Lockheed LETA is described in Section 5.1. The McDonnell Douglas LETA is not covered by this report. The remainder of both suction systems are described in Section 5.2.

5.1 LEADING EDGE TEST ARTICLE (LETA)

The LETA incorporates a configuration to provide for boundary layer air to be ingested through the surface in a measured and controlled manner. In addition, the leading edge provides liquid dispensing capability to prevent insect accretion that might cause transition on the leading edge, and to provide anti-icing capability. Instrumentation is included in the leading-edge test article to determine the extent of laminarization, the characteristics of the internal systems, and the surface pressure distribution.

The nominal suction requirements are defined for the leading-edge by aerodynamic analysis for the specific airfoil. The suction requirements for the LETA are defined in Section 7.6 in terms of a distributed suction. These distributed suction requirements are met by equivalent suction through discrete spanwise slots. Measurement and control of the suction distribution requires that each slot suction flow be individually ducted to the fuselage, where provision is made for flow measurement and control.

Provision is made to prevent insect accretion on the leading edge by covering the leading edge with a liquid film. This is achieved by ejecting liquid onto the surface through spanwise slots placed to provide film coverage over both the upper and lower airfoil surfaces for all normal excursions of attachment line movement. This liquid film also provides the anti-icing capability. These slots and the internal ducting are configured to provide adequate spanwise coverage during aircraft acceleration and pitch-up conditions during take-off and landing. Where cleaning liquid slots coincide with suction slots, special dual-purpose slots are provided to accommodate both functions. The dual-purpose slots are individually ducted from the LETA to interface with other LFC systems. This provides for control and measurement of the suction distribution and control of the liquid distribution where the two functions overlap. The two dedicated cleaning/anti-icing slots are manifolded together in the LETA and routed from the LETA by a single duct.

5.1.1 Slot Configuration

The slot configuration is the result of analytical evaluations consistent with the required distributed suction, physical limitations and slot aerodynamic limits. The resulting slot configuration was accordingly defined to be com-

patible with the taper of the leading-edge test article, manufacturing techniques, and internal structure.

Physical limitations established by currently available equipment for sawing slots through the titanium outer skin of the leading-edge surface dictate that the minimum nominal widths of the slots must be about 0.0094 cm (0.0037 in). A reasonable tolerance on slot width is acceptable, but efforts have been made to hold the tolerance to a minimum. Physical limitations also dictate that the slots must nominally be no closer than 1.57 cm (0.62 in). Reasonable tolerances on this spacing are acceptable. There are no physical upper limit constraints on slot widths or spacings.

5.1.1.1 Slot Locations

The suction slot locations were analytically established at the inboard end of the active section at wing station 141.14 for both the upper and lower surfaces, as in Tables 4 and 5 and Figure 117. The slot alphanumeric identification shown is used for all further reference to these slots. The locations of the six dual-purpose slots, D1 through D6, are chosen to be compatible with both the requirements for the location of the most forward suction slots in each surface and the requirements of the cleaning slots.

Dedicated cleaning slots were located between D1 and D2 in accordance with the 1.57 cm (0.62 in) minimum spacing, resulting in dedicated cleaning slots C1 and C2. These slots are connected to a common manifold tube inside the LETA as indicated previously.

Dedicated suction slots U1 through U8 on the upper surface, L1 through L6 on the lower surface, and dual-purpose slots D3 through D6 were located to satisfy the suction requirements but were limited by the minimum spacing permitted by physical limitations. The remaining upper and lower surface slots, U9 through U11 and L7 and L8, respectively, were located to satisfy the suction and structural requirements and the slot criteria limits. Slot criteria are discussed in Section 7.6.2.

TABLE 4. SLOT/DUCTING SYSTEM GEOMETRY-UPPER SURFACE

SLOT IDENT.	LOCATION x/c*	SLOT SPACING CN*		SLOT WIDTH W		MANIFOLD TUBE INSIDE DIAMETER		MANIFOLD ROUTING
		CM	IN	CM	IN	CM	IN	
C1	.00010	—	—	.0094	.0037	1.11	(7/16)	(COMBINED
C2	.00190	—	—	.0094	.0037	—	—	INBOARD)
D1	.00055	—	—	.0094	.0037	1.11	7/16	OUTBOARD
U1	.00269	1.57	0.62	.0094	.0037	1.11	7/16	OUTBOARD
U2	.00584	1.57	0.62	.0094	.0037	1.43	9/16	INBOARD
U3	.00967	1.57	0.62	.0094	.0037	1.27	1/2	INBOARD
U4	.01397	1.57	0.62	.0094	.0037	1.27	1/2	INBOARD
U5	.01857	1.57	0.62	.0094	.0037	1.27	1/2	INBOARD
U6	.02337	1.57	0.62	.0094	.0037	1.27	1/2	INBOARD
U7	.02829	1.57	0.62	.0094	.0037	1.27	1/2	INBOARD
U8	.03330	1.57	0.62	.0094	.0037	1.27	1/2	INBOARD
U9	.04951	4.95	1.95	.0102	.0040	1.27	1/2	INBOARD
U10	.08106	9.40	3.70	.0114	.0045	1.27	1/2	INBOARD
U11	.11299	9.40	3.70	.0127	.0050	1.27	1/2	INBOARD

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TABLE 5. SLOT/DUCTING SYSTEM GEOMETRY-LOWER SURFACE

SLOT IDENT.	LOCATION x/c*	SLOT SPACING CN*		SLOT WIDTH W		MANIFOLD TUBE INSIDE DIAMETER		MANIFOLD ROUTING
		CM	IN	CM	IN	CM	IN	
D2	.00516	—	—	.0094	.0037	0.79	5/16	OUTBOARD
D3	.00929	1.63	0.64	.0094	.0037	1.11	7/16	OUTBOARD
D4	.01384	1.63	0.64	.0094	.0037	1.11	7/16	OUTBOARD
D5	.01865	1.63	0.64	.0094	.0037	1.11	7/16	OUTBOARD
D6	.02363	1.63	0.64	.0094	.0037	1.11	7/16	OUTBOARD
L1	.02874	1.63	0.64	.0094	.0037	1.11	7/16	INBOARD
L2	.03395	1.63	0.64	.0094	.0037	1.11	7/16	INBOARD
L3	.03925	1.63	0.64	.0094	.0037	1.11	7/16	INBOARD
L4	.04466	1.63	0.64	.0094	.0037	1.11	7/16	OUTBOARD
L5	.05011	1.63	0.64	.0094	.0037	1.11	7/16	OUTBOARD
L6	.05562	1.63	0.64	.0094	.0037	1.11	7/16	OUTBOARD
L7	.07922	6.86	2.7	.0102	.0040	1.11	7/16	INBOARD
L8	.10711	7.87	3.1	.0114	.0045	1.11	7/16	INBOARD

*AT W.S. 141.14 ONLY

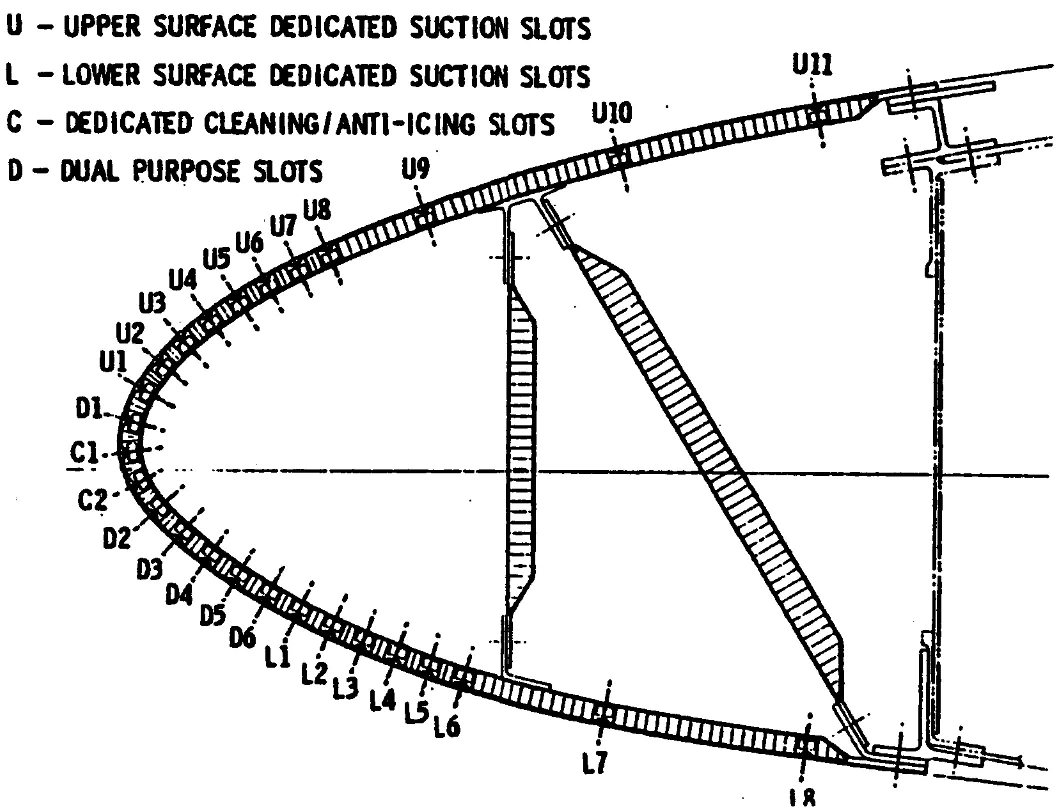

Figure 117. Leading Edge Slot Locations

The locations shown in Tables 4 and 5 and Figure 117 are the locations and nominal widths of the slots at wing station 141.14. The slot x-distance from the 0.0 x/c point on the airfoil is duplicated at the outboard end of the active section at wing station 188.078. The slots are maintained as straight lines on an equivalent unrolled surface and are at approximately a constant x-distance from the 0.0 x/c location at intermediate wing stations. Minor variations occur as a consequence of the requirements of manufacturing techniques.

5.1.2 Slot Ducting and Metering

Surface ducting is defined for this report as the slot ducting and collector ducting shown in Figure 118. Internal ducting is described later. The slot ducting and metering system is configured to provide stable slot flow and, where physically possible, to reduce spanwise flow maldistribution resulting from spanwise Cp gradients. The configuration is shown in Figure 118, and the dimensions are tabulated in Table 6. The configuration was selected to limit flow variations that result from discrete metering holes, while providing adequate structural continuity.

5.1.2.1 Slot Duct

Physical constraints were imposed on the slot duct to limit the titanium skin overhang to a minimum, but not to exceed 0.38 cm (0.15 in). The depth of the slot duct was limited by other geometric structural constraints. The slot is cut on the slot duct centerline. The selected slot duct is semicircular in cross section with the flat surface covered by the titanium skin. The slot duct radius is 0.152 cm (0.060 in). The slot ducts are the same for all slots.

5.1.2.2 Slot Duct Metering Holes

All slot duct metering holes, except those for slot U11, have 0.076 cm (0.030 in) diameters on 0.508 cm (0.2 in) centers, which correspond to the minimum physical limits. Slot U11 has metering hole diameters of 0.089 cm (0.035 in) on 0.508 cm (o.2 in) centers. All metering holes are drilled on the centerline of the slot duct and extend spanwise to the ends of the active (unblocked) slot.

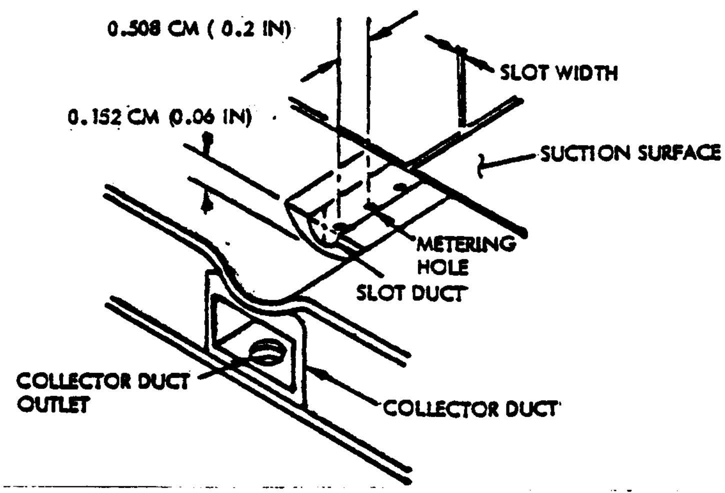

Figure 118. Slot/Metering/Ducting Configuration

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TABLE 6. METERING/DUCTING GEOMETRY-ALL SLOTS

SLOT DUCT	0.1520 CM (0.06 IN) RADIUS - SEMICIRCULAR	
METERING ORIFICE - DIAMETER	0.076 CM (0.030 IN) (EXCEPT 0.089 CM (0.035 IN) FOR U11)	
METERING ORIFICE - DEPTH	0.356 CM (0.14 IN)	NOMINAL
METERING ORIFICE - SPACING	0.508 CM (0.20 IN)	NOMINAL
COLLECTOR DUCT - WIDTH	0.787 CM (0.31 IN)	NOMINAL
COLLECTOR DUCT - DEPTH	0.406 CM (0.16 IN)	NOMINAL
COLLECTOR DUCT OUTLET - DIAMETER	0.478 CM (0.188 IN)	NOMINAL
COLLECTOR DUCT OUTLET - THICKNESS	0.203 CM (0.08 IN)	MINIMUM
COLLECTOR DUCT OUTLET - SPACING	15.24 CM (6.0 IN)	TYPICAL

5.1.2.3 Collector Ducts

The collector ducts serve as a manifold between the metering holes and the internal duct system to collect and communicate flow between the two. They serve to help overcome the effects of spanwise pressure gradients on slot flow by reducing the magnitude of pressure variations across the slot duct metering holes during suction. During the cleaning flow operation, they serve to distribute the cleaning flow spanwise. They limit excessive cleaning flow to the outboard end and ensure adequate flow to the inboard end during high accelerations and high pitch-up conditions, where the hydraulic head in the outboard ends of the system tends to increase significantly. All collector ducts are rectangular in cross section, having nominal dimensions of 0.787 cm (0.31 in) by 0.406 cm (0.16 in) with the 0.787 cm (0.31 in) dimension parallel to the local airfoil surface. All slot ducts and slots are filled at the inboard and outboard ends beyond the limits of the active section, at wing station 141.14 and 188.078, respectively, except as physical limitations may dictate some local variations. All collector ducts are blocked at these limits of the active section or at the ends of the test panel.

5.1.2.4 Dedicated Suction Slots

The collector ducts of all dedicated suction slots are uninterrupted for their full length. Outlet openings are located at 15.24 cm (6 in) intervals along the length of the active suction section to transfer the flow to the internal ducting. The outlet openings at either end of the duct are located as near to 7.62 cm (3 in) from the ends of the active section as physically possible. These outlet holes from the collector duct to the internal ducting are nominally 0.478 cm (0.188 in) inside diameter for all dedicated suction slots.

5.1.2.5 Dedicated Cleaning Slots

The collector ducts of dedicated cleaning slots have dams or blockages at 15.24 cm (6 in) intervals along the active section of the duct to limit spanwise liquid flow in the ducts. The slot ducts and slots are filled beyond the inboard and outboard ends of the active slot length. Cleaning flow is introduced to each of the 15.24 cm (6 in) collector duct segments through a restrictor. Each restrictor is 1.65 cm (0.65 in) long, with an internal diameter of 0.117 cm

(0.046 in) to 0.130 cm (0.051 in). The collector duct end of the opening is chamfered 0.785 rad (45 deg) to a face diameter of 0.279 cm (0.110 in). The restrictors provide a relatively high pressure loss to the cleaning liquid flow, thereby assuring relatively uniform flow into each of the 15.24 cm (6 in) segments regardless of the attitude or acceleration of the aircraft.

5.1.2.6 Dual-Purpose Slots

The dual-purpose slots incorporate the required features of the dedicated cleaning slots with provision for meeting the suction requirements. Dams are located in the collector duct as described for the dedicated cleaning slots. Modified check valves replace the restrictors and are located between the collector duct outlets and the internal ducting. The check valves used are Nupro Company CP-series check valves (P/N A-4CP2-1). The springs of these check valves have been removed. The valves are mounted directly to the internal manifold ducts and oriented with the poppet in the "closed" position toward the collector duct. These valves are "closed" when cleaning liquid is flowing to the slot from the internal ducts. For this case, the flow must pass through restrictor holes drilled through the poppets. During suction system operation, the valves are in the "open" position, providing low resistance to the suction flow.

5.1.3 Internal Ducting

Internal ducting is provided in the leading-edge test article to duct the suction flow from the collector duct outlets to the LETA interface and the cleaning flow from the interface to the collector duct, as applicable. All slots, except the dedicated cleaning slots, have a separate duct to permit individual measurement and control of suction flow and individual adjustment of cleaning flows. The dedicated cleaning slots, C1 and C2, have common ducting downstream of their restrictors in the suction direction. Typical internal ducting configurations are shown in Figure 119. All internal ducting lines are designed to withstand internal (burst) pressure of at least 172.4 KPa (25 psid).

5.1.3.1 Dedicated Suction Slots

The internal ducting of the dedicated suction slots is circular in cross-section and fabricated using aluminum tubing and rubber hose. The inside diameters of the piccolo or manifold tubes are shown in Tables 4 and 5, under the heading "Manifold Tube Diameter." These manifold duct diameters apply for all ducting in the leading-edge article extending to the inboard interface of the test article.

The manifold ducts for upper-surface dedicated suction slots U2 through U11 and L1, L2, L3, L7, and L8 are blocked at the outboard ends of the active section at approximately wing station 188.078. The inboards ends are routed to the inboard interface. Obstructions and bends in this ducting have been held to a minimum.

The manifold ducts for upper-surface slot U1 and lower-surface dedicated suction slots L4 through L6 are blocked at the inboard end of the active section at approximately wing station 141.14. The outboard ends are routed back through the leading-edge test article to the interface. Efforts have been made to avoid sharp bends and minimize obstructions to the flow path.

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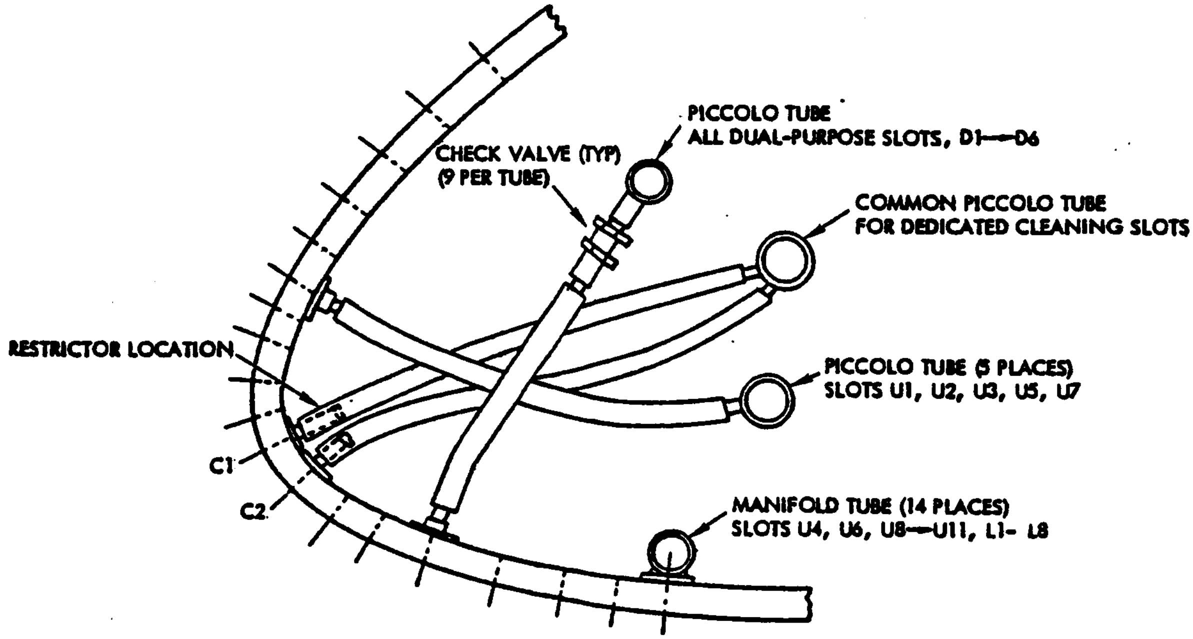

Figure 119. Internal Ducting Configurations

The minimum diameter of any connecting passages between the internal ducting and the collector duct is approximately 0.478 cm (0.188 in) for all dedicated suction slots. These connections do not infringe on the cross-sectional areas of either the collector ducts or the manifold ducts.

5.1.3.2 **Dedicated Cleaning Ducts**

The internal ducting of the dedicated cleaning slots consists of a common piccolo tube having a minimum internal diameter as shown in Table 4. This duct is plumbed to the protruding connector/restrictor at the inner surface of the test article shell by flexible air-hose as described in Section 4.1.2. The minimum diameter of the ducting between the restrictor/connector and the piccolo tubes is 0.478 cm (0.188 in). The duct is blocked at the outboard end wing station 188.078, and the inboard end is routed to the interface of the leading-edge test article.

5.1.3.3 **Dual-Purpose Slots**

The internal ducting of the dual-purpose slots is of the piccolo tube configuration and have internal diameters as indicated in Tables 4 and 5. The check valves described previously are mounted directly to the manifold tubes. The other ends of the check valves are connected to the collector duct outlet with ducting having a minimum inside diameter of 0.478 cm (0.188 in). The piccolo tubes of all dual-purpose slots are blocked at their inboard ends at approximately wing station 144.14. The outboard ends are routed back through the leading-edge test article to the inboard interface. The inside diameters of all internal ducting for the dual-purpose slots remain unchanged to the inboard interface. The routing is made to minimize sharp turns and obstructions to the flow.

5.2 SUCTION SYSTEM

The Lockheed suction system has been designed to provide maximum suction capability with respect to the design suction distribution over the upper and lower surfaces of the leading-edge test article within the physical constraints of the flight test aircraft. Sufficient flexibility has been included to permit alteration of the suction distribution to the extent that the flow of any slot or slots may be independently varied from 0 to 150 percent of the nominal design value with a design Cp distribution over the leading-edge surface. In addition, the system provides suction for two dedicated cleaning/anti-icing liquid slots to prevent outflow during laminar flow testing. The locations and identification of the Lockheed slots are shown in Figure 117. This system interfaces with the Lockheed console, the turbocompressor suction system, and the cleaning/anti-icing and purge systems. These interfaces will be described later.

The corresponding McDonnell Douglas suction system is similar to the Lockheed system, except that it applies only to the upper surface and there is no direct interface between the cleaning anti-icing systems and the suction system. The McDonnell Douglas suction system ducting has been designed within the same physical constraints as the Lockheed system to provide maximum suction capability. This design is based on suction flows and pressures specified by McDonnell Douglas. The McDonnell Douglas system has a requirement for a clearing system to eliminate cleaning liquid from the leading-edge test article suction ducting. The clearing system interface is common to the Lockheed purge system.

5.2.1 Lockheed Suction System Ducting

This section includes the interconnection between the individual slot-manifold duct lines of the Lockheed leading-edge test article to the chamber valves as illustrated in Figure 120. Required intermediate interconnections with the cleaning/anti-icing system are indicated. All lines and components of this system are compatible with the maximum operating system pressures: an internal pressurization to 172.4 KPa (25 psig) and evacuation to -82.7 KPa (-12 psig). Fittings and components have been selected or modified so that their equivalent flow areas are no smaller than that of the smallest line to which they are directly connected. Abrupt or large changes in flow area have been avoided. All components of the system have been selected to minimize line length and the number of sharp bends. All components in the system are suitable for use with propylene glycol methyl ether (PGME).

5.2.1.1 Leading-Edge Slot Lines

Leading-edge slot lines are connected to the leading-edge test article manifold lines at the inboard interface with the test article. When a change in line size occurs at the interface, a special adapter has been installed. This adapter consists of a conical diffuser having a total included angle of 0.140 rad (8 deg) or less. The purpose of this adapter is to minimize flow disturbances at the transition and thus internal noise which might interfere with laminar flow on the test article surface. The individual slot lines pass through the inboard wing leading edge. The dedicated suction slot lines connect directly to a pressure bulkhead fitting located in the leading-edge wing root at the shell of the fuselage pressure vessel. The dedicated cleaning/anti-icing

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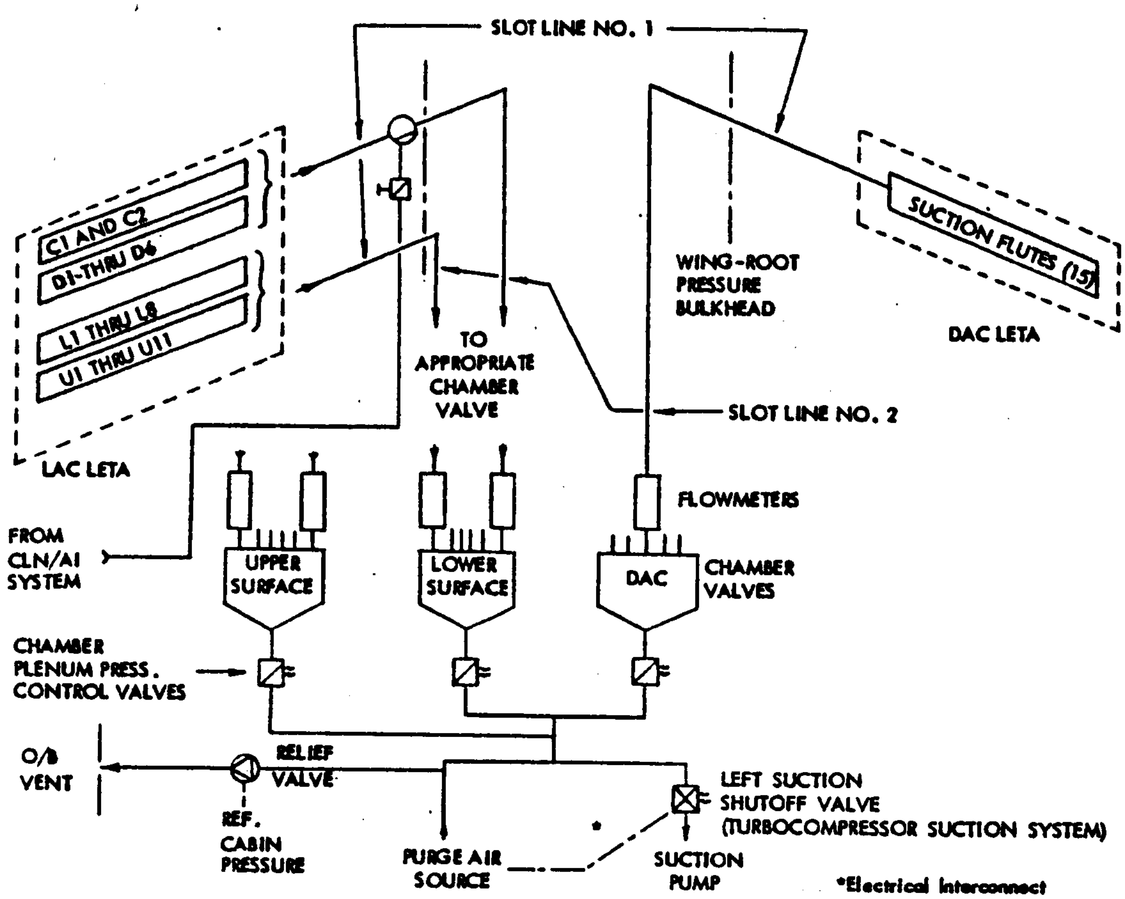

Figure 120. Suction System Schematic

slots, C1 and C2 in Figure 117, and dual-purpose slots, D1 through D6, are similarly routed to the wing root area, where they interface with both the cleaning/anti-icing system and the suction system shown in Figure 120. C1 and C2 are manifolded together inside the test article so that one slot line serves both. There are a total of 26 slot lines. The line sizes are shown as Line No. 1 in Table 7.

The interface of the cleaning/anti-icing and the dual-purpose slot lines with the cleaning/anti-icing and the suction systems provides porting of the slot lines to either system.. This porting is provided by a 3-way valve located in each individual line from the test article and connecting it to corresponding individual lines for suction and for cleaning/anti-icing. These 3-way valves are remotely operable from a single switch on the operator's control console.

Pressure instrumentation is to be provided for slots U2, U9, U4, U6, and L5.

5.2.1.2 Wing Root Pressure Bulkhead

All fittings at the leading-edge wing root bulkhead have been sized in accordance with the lines selected for the leading-edge interconnecting lines

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TABLE 7. LOCKHEED SUCTION LINE SIZES

SLOT NO.	FLOW*		(1) LINE 1 (O.D.)		(2) LINE 2 (O.D.)	
	KG/SEC.	LB/SEC.	CM	IN	CM	IN
C1/C2		—	1.588	.625	1.588	.625
D1	0.001085	.002392	1.270	.500	1.270	.500
U1	.001283	.002828	1.588	.625	1.588	.625
U2	.001507	.003323	1.588	.625	1.588	.625
U3	.001458	.003214	1.588	.625	1.905	.750
U4	.001986	.003275	1.588	.625	3.810	1.500
U5	.001498	.003302	1.905	.750	1.905	.750
U6	.001504	.003316	1.905	.750	3.810	1.500
U7	.001516	.003342	1.905	.750	3.810	1.500
U8	.001500	.003308	2.223	.875	2.223	.875
U9	.001509	.003327	2.223	.875	2.540	1.000
U10	.001993	.003292	2.223	.875	2.540	1.000
U11	.001428	.003148	1.905	.750	2.540	1.000
D2	.001115	.002458	1.588	.625	1.588	.625
D3	.001580	.003484	1.588	.625	1.588	.625
D4	.001605	.003539	1.588	.625	1.588	.625
D5	.001622	.003575	1.588	.625	1.588	.625
D6	.001624	.003581	1.588	.625	1.905	.750
L1	.001717	.003786	1.588	.625	1.588	.625
L2	.001705	.003759	1.588	.625	1.588	.625
L3	.001688	.003721	1.588	.625	1.588	.625
L4	.001640	.003616	1.905	.750	1.905	.750
L5	.001619	.003570	1.588	.625	3.810	1.500
L6	.001604	.003537	1.588	.625	3.810	1.500
L7	.001466	.003231	1.588	.625	1.588	.625
L8	.001486	.003276	1.588	.625	1.588	.625

* LINES SIZED FOR THE BASELINE FLIGHT CONDITION:

11,582 M (38,000 FT) ALT, M_N = 0.75 AT 50% OVER NOMINAL SUCTION LEVEL.

(1) LINE FROM TEST PANEL TO FUSELAGE
(2) LINE FROM FUSELAGE INTERFACE TO CHAMBER VALVE.

NOTE: LINE DIAMETERS ARE NOMINAL OUTSIDE DIAMETERS AND ASSUME 0.089 CM (0.035 INCH) WALL THICKNESS FOR TUBES 2.54 CM (1.000 INCH) OD AND LARGER; FOR SMALLER TUBES 0.071 CM (0.028 INCH) WALL THICKNESS IS ASSUMED

and provide the transition to the fuselage line sizes listed as Line 2 in Table 7. These fittings do not restrict the flow areas below those of the interconnecting lines. Standard fittings have been used where practical.

5.2.1.3 Fuselage Interconnecting Lines

The fuselage interconnecting lines route the individual slot suction system flows from the wing root pressure bulkhead area to the suction chamber valves. These lines are routed to minimize the length and number of bends. Sharp bends have been avoided where practical. The routing avoids areas where the lines would be exposed to heat sources significantly higher than cabin temperature at cruise conditions; however, the lines are thermally insulated. The lines are routed as continuous lines from the pressure bulkhead to the flowmeters or flowmeter adapters. The diameters of these lines are given as Line No. 2 in Table 7.

Provision has been made for the mounting of a Hastings Flowmeter in any of the fuselage interconnecting lines. The flowmeters attach to the slot line with a conical diffuser and reducer at its inlet and discharge ends, respectively. Since each slot line must be capable of accepting a flowmeter and a maximum of six flowmeters are used, each line must have an adapter installed when the flowmeter is not used.

5.2.1.4 Chamber Valves

The Lockheed suction system includes two chamber valves as illustrated in Figure 120. The chamber valves and control electronics were designed and fabricated at NASA Langley. One chamber valve connects primarily to the upper-surface slots and one to the lower-surface slots. The upper-surface chamber valve is connected to slots C1 and C2 (paired), D1, and U1 through U11 for a total of 14 slots and 13 slot lines. The lower-surface chamber valve is connected to slots D2 through D6, and L1 through L8 for a total of 13 slots and slot lines. Each chamber valve is provided with a sonic needle valve for each of these slot lines. The exits of both chamber valves connect to a 8.89 cm (3.5 in) OD duct. These chamber valves are located with the lower surface chamber valve in the forward position.

The chamber valves are designed for an operating internal evacuation of 82.7 KPa (12 psid) pressure differential (crushing) and an operating internal pressurization to 103.4 KPa (15 psid) pressure differential (bursting). These values do not include any safety margin. The chamber valve plenum is thermally insulated for a 37.8°C (100°F) temperature differential between the inner and outer surfaces to prevent outer surface temperatures below 15.6°C (60°F) with a 12.9°C (75°F) cabin temperature.

All three chamber valves are of a single basic configuration adapted to satisfy the specific McDonnell Douglas or Lockheed requirements. Sonic needle valves are to be selected from several available sizes to best match each slot line requirement, i.e., diameter, flow, and pressure loss. Provision is made for installing up to 15 needle valves in the end plate of each chamber valve. Where less than 15 needle valves are required, the extra end plate openings are blanked off. All needle valves are used in conjunction with a common valve actuator configuration. The needle valve actuators are configured to produce an

approximately coaxial annular area at all travels. Each needle valve incorporates a position sensor as an integral part of its actuator. The individual valves are controlled from the consoles by an integral digital readout, knob, and potentiometer.

5.2.2 McDonnell Douglas Suction System Ducting

This section includes the interconnecting lines between the individual flutes of the McDonnell Douglas LETA and chamber valve as illustrated in Figure 120. All lines and components of this system have been selected for an operational internal pressurization of 172.4 KPa (25 psig) and evacuation to -82.7 KPa (-12 psig). There are no restrictions in fittings or components that result in equivalent flow areas smaller than that of the smallest line to which the fitting is directly connected. Abrupt or large changes in flow area and sharp turns have been avoided. All components of the system have been selected to minimize line length and the number of sharp bends. All components in the system are suitable for use with propylene glycol methyl ether.

5.2.2.1 Leading-Edge Flute Lines

Leading-edge flute lines connect to the McDonnell Douglas flutes at the inboard interface of the McDonnell Douglas leading-edge test article. These lines route the suction flow from the 15 McDonnell Douglas flutes, numbered from front to rear, to the wing root area at the shell of the fuselage pressure vessel. Where a change in size of a suction line occurs at the interface, concical diffuser-type adapters are used as for the Lockheed LETA to minimize noise generation. The lines are free of unnecessary bends and fittings and have line diameters as shown in the column labeled "Line 1" in Table 8.

5.2.2.2 Wing Root Pressure Bulkhead

All fittings at the leading-edge wing root bulkhead have been sized in accordance with the lines selected for the leading-edge interconnecting lines and provide the transition to the fuselage line sizes listed as Line 2 in Table 8. These fittings do not restrict the flow areas below those of the interconnecting lines. Sharp turns and abrupt changes in flow area have been avoided where practical. Standard fittings are used when practical.

5.2.2.3 Fuselage Interconnecting Lines

The fuselage interconnecting lines route the individual flute suction system flows from the wing root pressure bulkhead area to the suction chamber valve. These lines have been routed to minimize the length and number of bends. Sharp bends have been avoided. The routing avoids areas where the lines would be exposed to any heat source significantly higher than cabin temperature at cruise conditions, however thermal insulation is applied to the lines. The lines are routed as continuous lines from the pressure bulkhead to the chamber valve, except for the flowmeters or the flowmeter adapter where flowmeters are not installed. The line diameters are shown in the column headed "Line 2" in Table 8.

5.2.2.4 Chamber Valve

The McDonnell Douglas flute suction lines are connected to a chamber valve provided with a sonic needle valve for each flute.

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5.2.3 Pump Interconnection

Each chamber valve discharge is connected by a 8.89 cm (3.5 in) OD duct through individual chamber plenum pressure control valves to a common 12.7 cm (5 in) OD duct as shown schematically in Figure 120. The chamber plenum pressure control valves are slow-acting, remotely-positioned, butterfly type valves. A valve position sensor is an integral part of each valve. The 12.7 cm (5 in) OD common duct carries the total suction flow into the unpressurized aft compartment to the LEFT suction shutoff valve. The suction shutoff valve is a slow-acting, remotely-controlled, normally-closed butterfly type valve. From this valve, the system is connected to the suction pump and control system.

TABLE 8. DOUGLAS SUCTION LINE SIZES

FLUTE NUMBER	FLOW* (KG/SEC)	TOTAL PRESSURE @EXIT* (KPa)	(1) LINE/ (O.D.-CM)	(2) LINE 2 (O.D.-CM)
1	0.00306	25.63	1.905	1.905
2	0.00523	20.55	2.540	2.540
3	0.00445	16.75	2.540	3.175
4	0.00325	15.01	2.540	2.540
5	0.00336	13.52	2.540	3.175
6	0.00371	12.82	3.810	3.810
7	0.00318	12.60	3.175	3.175
8	0.00203	12.78	1.905	2.540
9	0.00156	12.86	1.905	1.905
10	0.00161	12.73	1.905	1.905
11	0.00175	12.65	1.905	1.905
12	0.00175	12.69	1.905	1.905
13	0.00175	12.65	1.905	1.905
14	0.00175	12.69	1.905	1.905
15	0.00175	12.69	1.905	1.905

* BASED ON REVISED McDONNELL DOUGLAS FLOW DONCITION AT INTERFACE (31 MARCH 1982); BASELINE SUCTION (ALT = 11.582 CM (38,000 FT), M = 0.75) @ 150% FLOW

(1) LINE FROM TEST PANEL TO FUSELAGE

(2) LINE FROM FUSELAGE INTERFACE TO CHAMBER VALVE

NOTE: LINE DIAMETERS ARE NOMINAL OUTSIDE DIAMETERS AND ASSUME 0.089 CM (0.035 IN) WALL THICKNESS FOR TUBES OF 2.54CM (1.000 IN) OD AND LARGER; FOR SMALLER TUBES 0.071 CM (0.028 IN) WALL THICKNESS IS ASSUMED

TABLE 8. DOUGLAS SUCTION LINE SIZES (CONT'D)

FLUTE NUMBER	FLOW* (LB/SEC)	TOTAL PRESSURE @ EXIT* FROM TEST PANEL (PSFA)	LINE 1 (1) (O.D.-IN)	LINE 2 (2) (O.D.-IN)
1	.00674	535.2	0.750	0.750
2	.01152	429.1	1.000	1.000
3	.00981	349.8	1.000	1.250
4	.00717	313.4	1.000	1.000
5	.00741	282.4	1.000	1.250
6	.00819	267.8	1.500	1.500
7	.00702	263.2	1.250	1.250
8	.00447	266.9	0.750	1.000
9	.00344	268.6	0.750	0.750
10	.00354	262.8	0.750	0.750
11	.00386	264.1	0.750	0.750
12	.00386	265.1	0.750	0.750
13	.00386	264.1	0.750	0.750
14	.00386	265.1	0.750	0.750
15	.00386	265.1	0.750	0.750

* BASED ON REVISED McDONNELL DOUGLAS FLOW CONDITION AT INTERFACE (31 MARCH 1982); BASELINE SUCTION (ALT = 38,000 FT, M = 0.75) @ 150% FLOW

(1) LINE FROM TEST PANEL TO FUSELAGE

(2) LINE FROM FUSELAGE INTERFACE TO CHAMBER VALVE

NOTE: LINE DIAMETERS ARE NOMINAL OUTSIDE DIAMETERS AND ASSUME 0.035 INCH WALL THICKNESS FOR TUBES OF 1.000 INCH OD AND LARGER; FOR SMALLER TUBES 0.028 INCH WALL THICKNESS IS ASSUMED.

A 6.35 cm (2.5 in) OD purge line tees into the 12.7 cm (5 in) OD common duct described above. This interface occurs inside the cabin area between the connections to the individual chamber valves and the LEFT suction shutoff valve. This line is connected to the purge system described in Section 5.5. Another 6.35 cm (2.5 in) OD line containing a pressure relief valve is teed into the 6.35 cm (2.5 in) OD line. The purpose of this relief valve is to prevent over-pressurizing the chamber valves or the purge air filter during purge operations. Any excess pressure is vented overboard.

The chamber plenum pressure control valves of both the Lockheed and McDonnell Douglas systems are remotely positioned from the control consoles. The LEFT suction shutoff valve and the purge system shutoff valves are controlled only from the turbocompressor control console (consolette) and are electrically connected to prevent the LEFT suction shutoff valve from being opened if either of the purge system shutoff valves is not fully closed. In addition, the LEFT suction shutoff valve is electrically interlocked with the Lockheed and McDonnell Douglas PGME/water liquid system shutoff valves of Section 5.4 to prevent the application of suction to the test articles if either of the liquid control valves is not fully closed.

5.3 SUCTION PUMP

5.3.1 Pump Selection

The LEFT installation on the JetStar requires a suction pump system which provides adequate suction flow for the McDonnell Douglas and Lockheed leading-edge test articles. The desired suction flows must be delivered within the physical constraints of the JetStar flight test vehicle, and each test article must be capable of being operated independently. The required suction pump must be compatible with the JetStar in terms of physical size and energy requirements. A schematic diagram of the pump system illustrating its interface with the suction system is presented in Figure 121.

The pump system selected for this application is a modified AiResearch Manufacturing Co. (AiResearch) ATCU 30 turbocompressor unit. This pump is used for cabin pressurization in the Boeing 707 aircraft. A schematic diagram of the unmodified pump is shown in Figure 122. The turbocompressor suction pump (TSP) system includes automatic controls and stall protection for operation of the suction pump system at altitude conditions, regardless of other suction flow control adjustments upstream of the pump system. Provision is incorporated for safe and stable operation of the pump for ground checkout conditions at a reduced pump pressure ratio. All associated controls, sensors, and servo systems required for operation of the turbocompressor in this operating mode are included as a part of the TSP.

5.3.2 AiResearch Turbocompressor Suction Pump (TSP)

The AiResearch Turbocompressor Suction Pump System (TSP) consists of the modified turbocompressor unit and all controls and protective systems required for safe pump operation in the LEFT application. This includes pump on/off controls, speed controls, overspeed shutdown protection, and regulation of pump pressure ratio. In addition, instrumentation required to monitor the pump operation is provided with the pump. All switches, selectors, controllers, sensors, etc., are provided with the system with the exception of the oil cooler and oil temperature controller. The basic required modifications to the pump are indicated in Figure 123. The left schematic indicates the pump configuration in the Boeing 707 cabin pressurization configuration. The right schematic illustrates the pump modification for the LFC suction operation with the surge vent valve moved from the ambient discharge vent of the Boeing installation to an ambient inlet vent for the suction operation. The turbine system remains virtually unchanged externally, although there are a number of additional modifications required that are internal to the turbine and to the built-in controls.

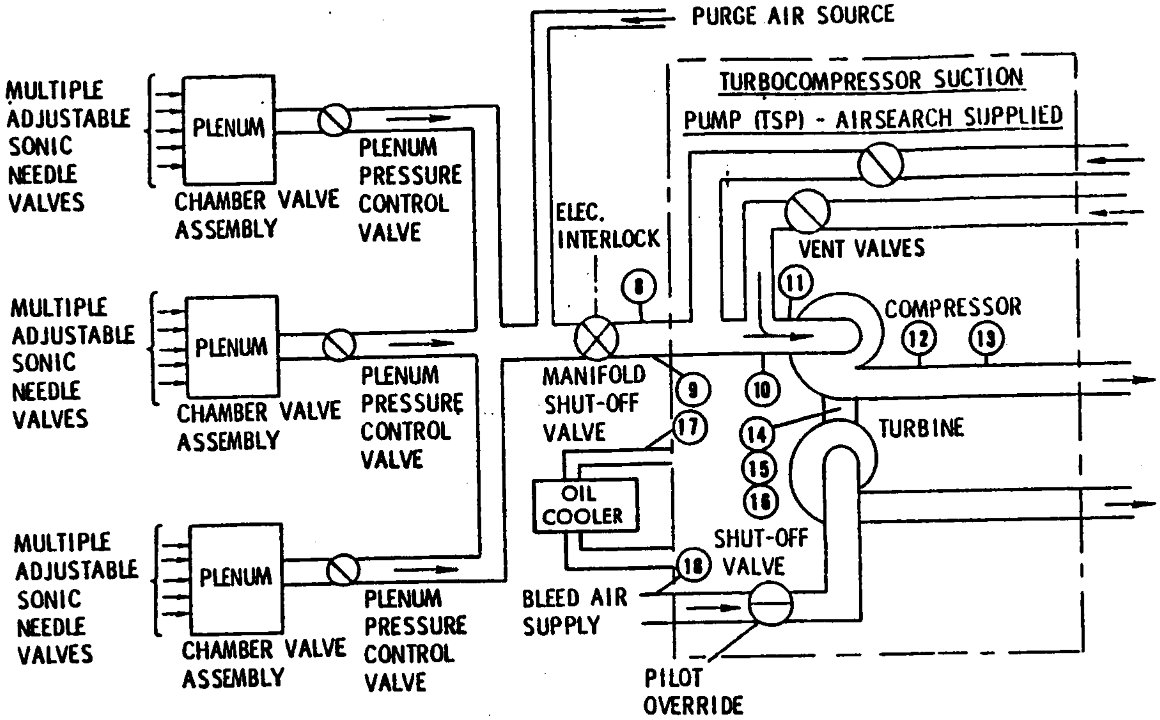

Figure 121. Suction Control System Schematic

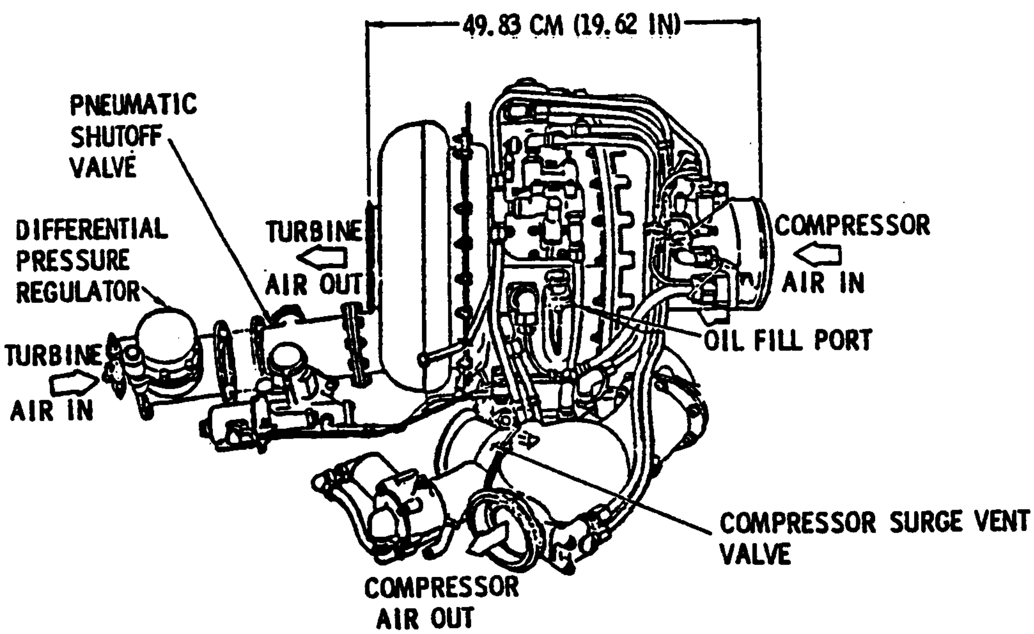

Figure 122. Turbocompressor Suction Pump (Unmodified)

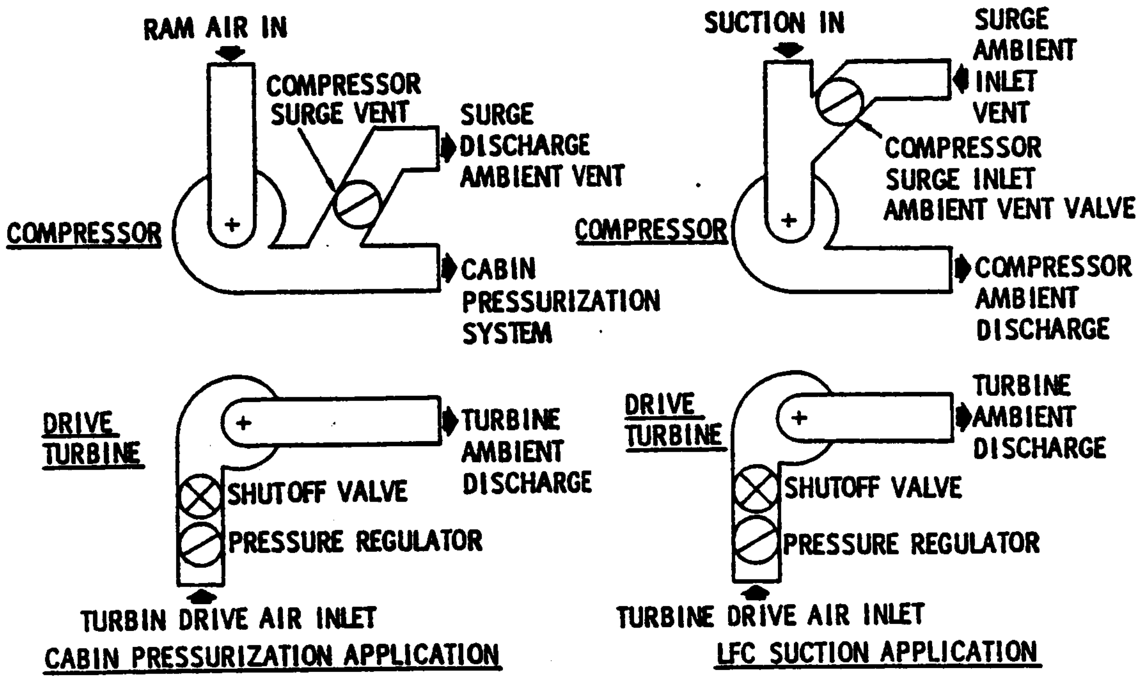

Figure 123. Basic Pump Modification

The TSP is capable of operation in its primary suction mode from 9144m to 13,716m (30,000 to 45,000 ft) flight pressure altitude and airplane flight Mach numbers from 0.70 to 0.80. During normal suction operation the plenum pressure control valves and the LEFT suction shutoff valve will be in the full-open condition, Figure 121. During purge operation, the LEFT suction shutoff valve will be full-closed and the chamber plenum pressure controlled by adjusting the plenum pressure control valve setting. The pump controls are all contained in the broken line rectangle at the right of Figure 121. All valves within this rectangle are to be provided by AiResearch with the shutoff valve on the bleed air supply to the turbine providing the primary operator pump control. The system also provides satisfactory intermediate operation for system checkout both statically on the ground, and if practicable, at lower flight altitudes and speeds subject to limitations specified by AiResearch. Air supply for the turbocompressor is provided by last-stage compressor bleed air extracted from the JetStar's four JT12A-6 engines. Both the turbine and suction pump elements of the turbocompressor discharge directly overboard to ambient pressure with an assumed 10 percent pressure drop.

5.3.2.1 TSP Performance Characteristics

The suction requirements for the Lockheed and McDonnell Douglas LEFT suction systems provide some flexibility of operation that may be used to aid in achieving the objectives of the test.

Figure 124 presents the pump performance characteristics for the LEFT application taken from AiResearch provided data.

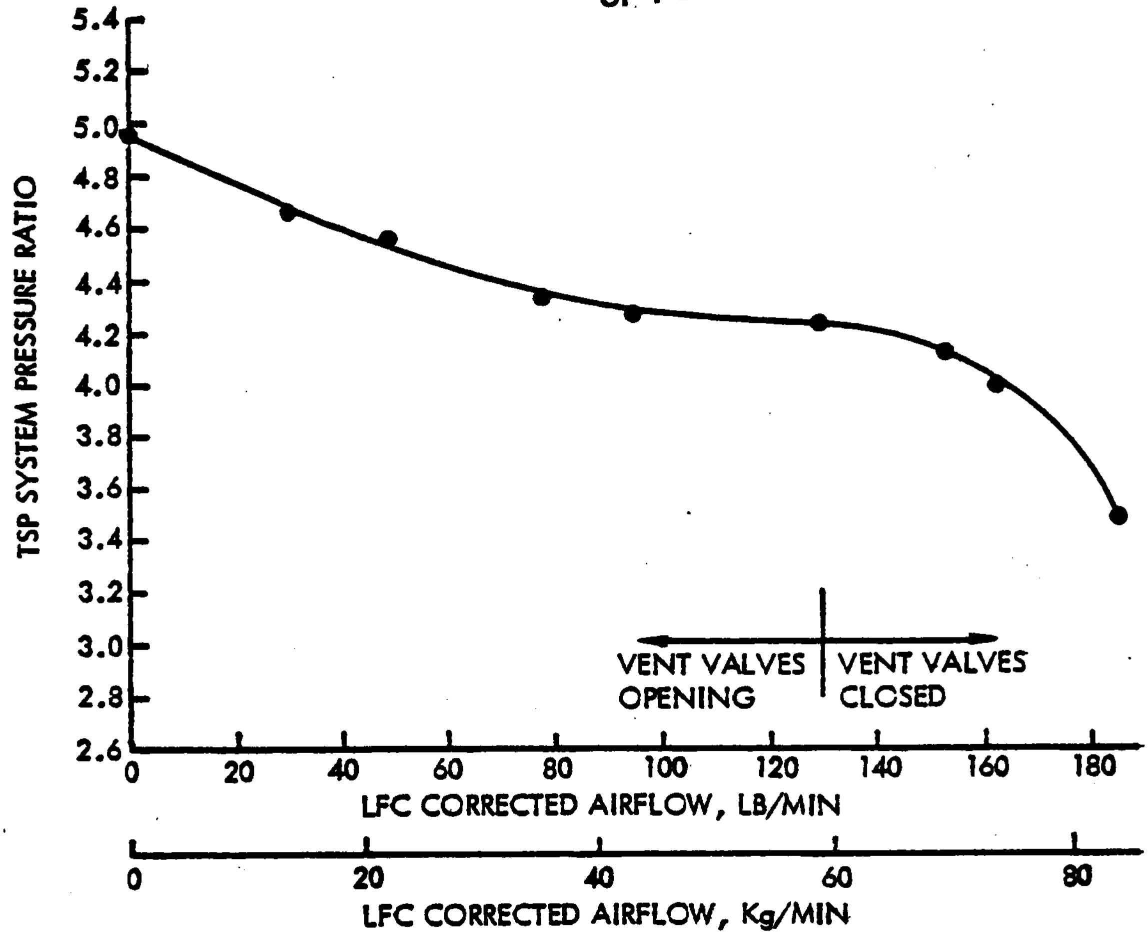

Figure 124. Pump Performance

The individual controls of the TSP system are described below. The purpose of these controls is to provide and maintain stable suction characteristics at the LEFT system/TSP interface with a minimum of attention and monitoring on the part of the operator. The overriding requirement of the TSP control system has been that it function as an integrated system, free from any oscillation, instabilities, or divergent characteristics throughout the required LEFT system operating envelope. The turbocompressor, its existing controls, and any additional controls have been modified to be consistent with the ultimate integrated turbocompressor control requirements.

Lockheed has designed or selected the upstream controls to meet the AiResearch specified response time requirements for compatibility with the TSP control systems.

All remotely mounted control components, servo systems, or electronic control sensors or controllers are provided with the TSP and external or remote control interconnections are electrical only.

Vent Valves

The vent valves are automatically controlled by the AiResearch provided TSP control systems to maintain the pump inlet corrected flow to a previously established schedule relative to the pump surge line as suction system flow is varied. This is accomplished by allowing larger or smaller quantities of ambient air to vent directly into the pump inlet. Under many operating conditions, the pump flow and pressure ratio will greatly exceed the requirements for the LEFT suction system requirements. In some cases, virtually all of the pump inlet flow must be provided through the vent valves. It is also intended that the LEFT systems will be completely shutoff during pump start-up. During this condition the vent valves will be in the full-open position.

The vent valves are designed to operate with sufficient rapidity to prevent any operation of the upstream LEFT control systems, including shutoff, to cause any sustained instabilities, oscillations, or objectionable compressor stall. Pump inlet pressure stability is reached within 10 seconds or less following any LEFT system control adjustments and the pressure ratio is expected to be maintained within 5 percent of the absolute value required by the pump operating characteristic. Lockheed has designed or selected upstream controls to be compatible with the response time of the TSP surge control (vent valve) system.

Turbine Bleed Air and Speed Control

The TSP includes a bleed-air shutoff valve and restrictor for the bleed-air supplied to the drive turbine of the TSP system. The turbine speed control system operation is compatible with the TSP performance characteristics, the vent valves, and the LEFT suction system requirements.

The control is designed to provide stable operation, free of any instabilities or oscillations, and to automatically limit the speed to the safe operating limit of the turbocompressor. The system is also designed to respond rapidly and with sufficient margin to prevent excessive or unsafe momentary overspeed conditions resulting from actuation of any valve in the LEFT or vent valve control systems. The TSP controls return the system to stable operation within 10 seconds or less. The system includes all equipment required to sense and compare actual rotation speed with the desired value and to automatically adjust any controls, whether internal or external to the turbocompressor, to return the speed to the desired value. This system is incorporated in the as-shipped TSP. The system incorporates either fail-passive or fail-active arrangements in all of its components and controls for the purpose of minimizing occurrence of Category I (catastrophic) and Category II (critical) failure modes as defined in MIL-STD-1629A. All control systems not self-contained in or mounted directly on the TSP assembly are electrically controlled.

General Control Features

A remote control panel is mounted directly on the McDonnell Douglas operator's console. This panel includes all switches, controls, and indicators required for safe operation of the TSP including special controls required for ground checkout operations. This panel is small enough to facilitate integration with the operator's console and any bulky electronic controls or other devices are capable of mounting remotely from this control unit.

5.3.2.2 TSP Design Considerations

Strength Design Point

The turbocompressor and all TSP system components are designed for safe operation throughout the range of system operating requirements. The bleed-air system components and the turbocompressor are designed for safe operation up to the maximum conditions resulting from the available bleed airflow-pressure-temperature conditions of the NASA JetStar. The design and materials of all selected components of the system are suitable for the maximum bleed pressure and temperature characteristics of the JT12A-6 engines.

Design Life

The TSP system components and all modifications thereto are suitable for a minimum operating service life of 1500 hours. All system components are consistent with continuous operation of the TSP for periods up to 2 hours in duration. Periodic maintenance of the system components is the same as their original procurement and is not impaired by any modifications or by their application.

Design Loads

The pump is designed to withstand the required limit load factor without experiencing any detrimental deformation or permanent set when subjected to limit load. The pump is also designed to withstand ultimate loads without failure. Ultimate loads are limit loads multiplied by a factor of 1.5.

Other Factors

The basic TSP with all modifications and added components is compatible with or meets specified requirements with respect to the following design requirements: environmental conditions, electrical power, reliability, maintainability, and transportability.

Lubrication System

The turbocompressor incorporates a self-contained lubrication system except for oil cooling requirements. Fill, drain, and measurement provisions are readily accessible. Controls or components that use lubrication oil in their operation are mounted directly on the turbocompressor and the plumbing for these components is incorporated on the TSP. The system is capable of continuous operations at any point in the suction envelope for a minimum of three 2-hour periods between servicing. The only connections in the system external to the TSP are in the inlet and discharge connections for the oil cooler. An oil cooler that satisfies the TSP system requirements and is suitable for installation in the NASA JetStar has been selected/designed by Lockheed. This oil cooler meets the pump requirements for heat rejection, pressure drop, and other characteristics established by AiResearch.

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5.4 CLEANING/ANTI-ICING SYSTEM

The Lockheed and McDonnell Douglas systems are entirely independent of one another and differ significantly in basic concepts. Both systems use a PGME/water solution as the insect protection liquid, and all components used in both systems are suitable for use with the PGME liquid. Due to the nature of the PGME liquid, all components of these systems containing the PGME liquid are located outside the passenger compartment. The inclusion of components of the systems inside the passenger compartment would require a more complex system and precautionary measures. Therefore, these liquid systems are confined to the aft compartment, the wing fairing, and the wing leading edge. All components containing the PGME liquid mixture are located in a well ventilated area with sufficient drainage provision to prevent entrapment of any liquid leakage. The system is designed to prevent a flammable mixture of PGME and air from forming in the vicinity of any ignition source in the event of any system leaks. For convenience, some non-liquid containing components of the electrical, pneumatic, and nitrogen systems that support the liquid systems are located in the passenger compartment.

The Lockheed system, Figure 125, uses selected slots around the leading edge to discharge a small flow of liquid onto the surface. This liquid forms a protective film over the surface to prevent insect accretion and prevent/remove ice buildup. The design requirements for this liquid flow have been demonstrated to provide a liquid film sufficient to prevent accretion in a high density insect environment.

The McDonnell Douglas contamination avoidance and anti-ice systems are totally independent of the comparable Lockheed systems except that both interface with the nitrogen pressurization system. The design and operation specifications for these systems are the responsibility of McDonnell Douglas. The contamination avoidance system is supported by Lockheed designed subsystems that meet requirements defined by McDonnell Douglas at the Lockheed/McDonnell Douglas interface.

The primary McDonnell Douglas system uses a slat deployed in front of the leading edge to act as a shield to deflect insects from the leading edge. In addition, a contamination-avoidance spray system, Figure 126, is used to apply a liquid onto the wing surface to provide additional protection from insect accretion. The spray nozzles of this system are located on the aft side of the shield. McDonnell Douglas anti-icing protection for the shield is provided by a separate system, Figure 127. The shield is protected from ice buildup by a TKS LTD system that oozes an ethylene glycol mixture onto its surface, forming a film over the shield. No components are used in either system that are sensitive to the TKS fluid mixture, which includes ethylene glycol, alcohol, and water.

5.4.1 Lockheed Cleaning/Anti-Icing System

The Lockheed system, illustrated schematically in Figure 125, interfaces with the suction system, the purge system, the nitrogen pressurization system, and the Lockheed console. The major components of the Lockheed system are the government furnished liquid supply tank and the fluid distribution system. This system delivers the 60 percent PGME/40 percent water mixture through the seven 3-way valves at the interface with the suction system to the eight cleaning/

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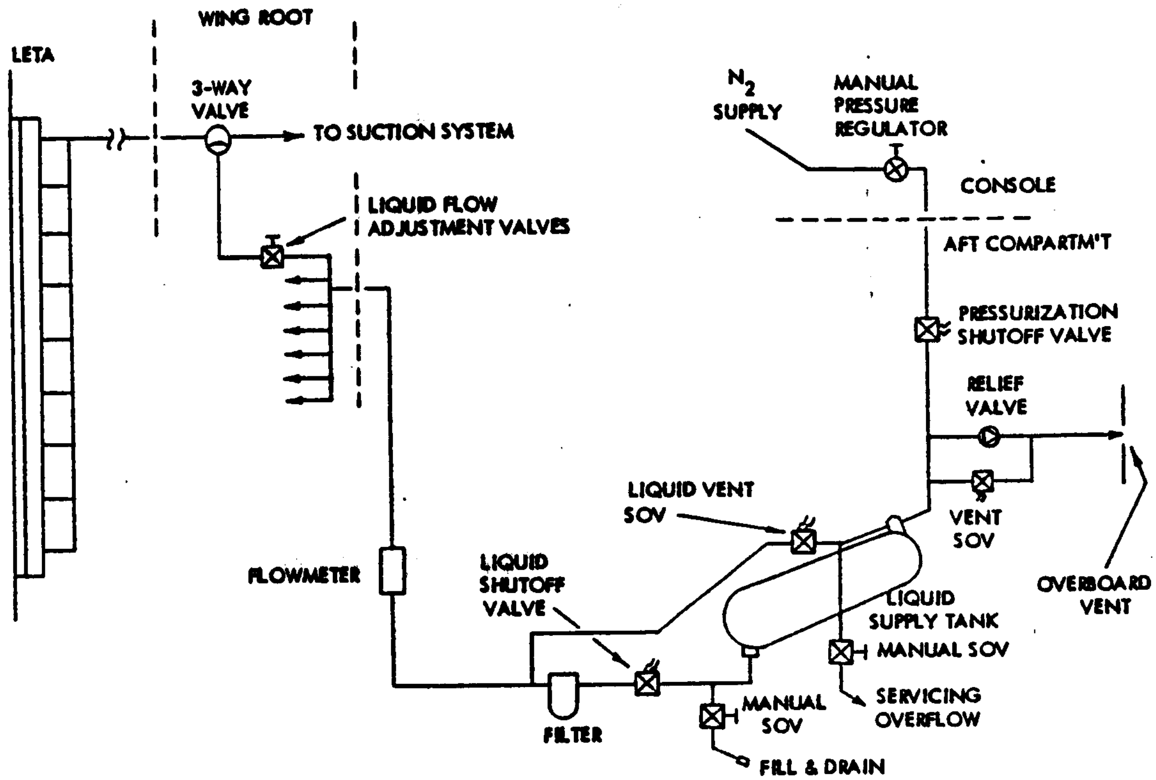

Figure 125. Lockheed Cleaning/Anti-Icing System

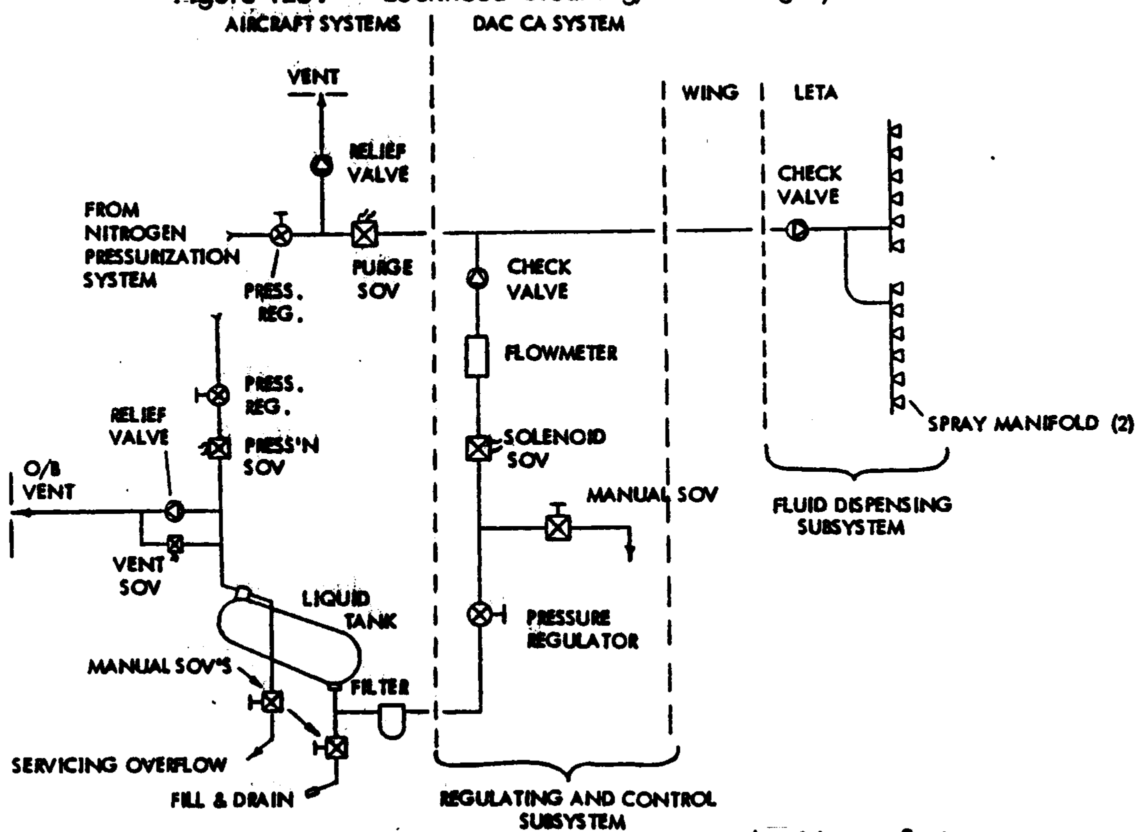

Figure 126. McDonnell Douglas Contamination Avoidance System

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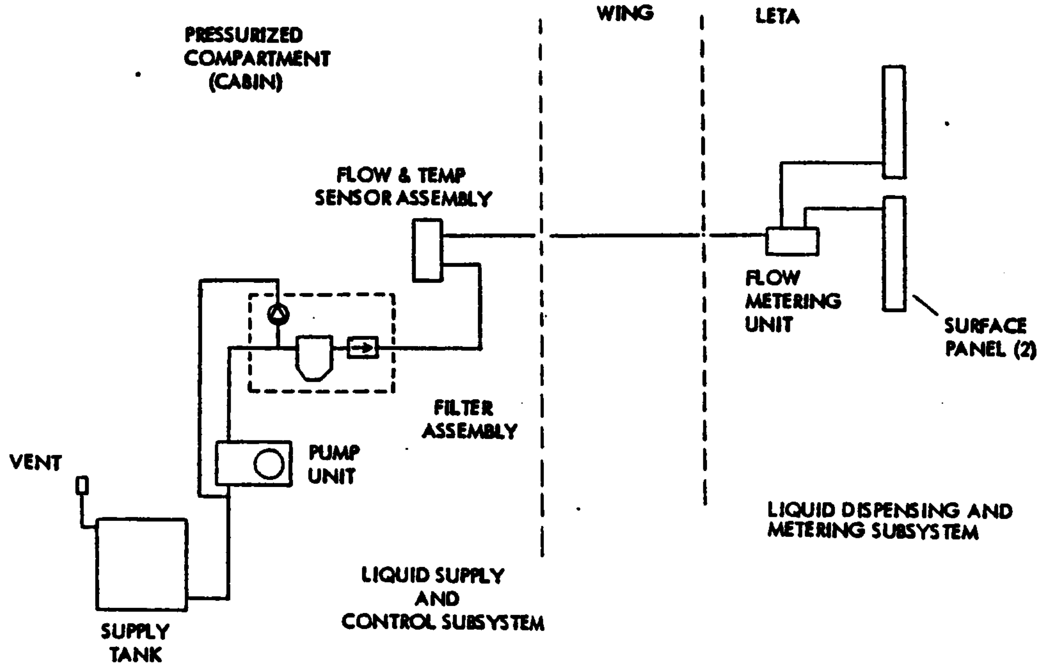

Figure 127. McDonnell Douglas Krueger Shield Ice Protection System

anti-icing slots. These slots (two dedicated cleaning slots and six dual-purpose slots) are located near the test section leading edge. Liquid flow control is provided by adjustment of the supply tank nitrogen pressurization pressure. Flow distribution is regulated by ground adjustable throttling valves located upstream of each 3-way valve.

The supply tank is installed so that liquid is supplied from the bottom of the tank through the single port located near one end of the tank. The two-port connector at the other end of the tank provides for pressurization/venting and for liquid return/servicing overflow as illustrated in Figure 125. The Lockheed cleaning/anti-icing system interfaces with the nitrogen pressurization system upstream of the manual pressure regulator. This nitrogen pressure regulator is located on the operator's console. The pressure regulator is connected to the nitrogen pressurization shutoff valve. Downstream of the shutoff valve, the vent/pressurization line connects to an overboard vent through parallel lines. One of these lines contains a pressure relief valve and the other a vent shutoff valve. The overboard vent consists of a scarfed vent mast so located that any liquid emitting from the mast will not blow back on any aircraft components. The nitrogen shutoff valve is connected to the smaller port of the two-port connector of the supply tank.

The lower port on the tank connects through the liquid shutoff valve and filter to the flowmeter. The line between the liquid filter and flowmeter is connected to the remaining port near the top of the supply tank through a fluid return line containing the liquid vent shutoff valve. A tank fill and drain line tees into the fluid supply line between the tank outlet and the liquid shutoff valve. This line provides for servicing of the tank with the PGME/water solution or for tank draining, and contains a manual shutoff valve. The fluid

return line likewise connects through a tee to a manual shutoff valve to provide an overflow for tank servicing. This connection is made between the liquid vent shutoff valve and the tank. For both manual shutoff valves described above, provision is made for draining PGME fluid clear of the aircraft. All components of the system between and including the nitrogen pressurization shutoff valve and the flowmeter are located in the unpressurized aft fuselage compartment. The liquid is plumbed from the flowmeter through the wing root fairing to a manifold plenum located in the left wing leading-edge root area. The manifold is provided with ports for connecting seven slot cleaning lines. These seven slot lines are routed from the manifold to manual liquid flow adjustment valves also located in the wing root area. These valves are accessible on the ground between flights. Each valve is connected to the corresponding slot suction line at the 3-way suction/cleaning selection valve located in the wing root. The 3-way valves provide a mechanical interlock to avoid inadvertent delivery of cleaning liquid to the chamber valves. All remotely operable valves are controlled from the operator's console for the Lockheed system.

5.4.2 McDonnell Douglas Contamination Avoidance System

The McDonnell Douglas contamination avoidance system includes the deployment of the shield to deflect insects and other contaminants from the active suction surface during operation at insect populated altitudes. Additional protection is provided by spraying a PGME/water solution over the active suction surface from nozzles located in the aft side of the deployed deflector. This section describes the latter, active liquid system shown in Figure 126.

The McDonnell Douglas contamination avoidance spray system interfaces with the government furnished fluid supply subsystem and nitrogen pressurization system. The fluid supply subsystem provides the PGME/water mixture to the McDonnell Douglas system at 2,068 +103.4/-0 KPa (300 +15/-0 psig). All components of the McDonnell Douglas contamination avoidance (CA) system are provided by McDonnell Douglas. The McDonnell Douglas system consists of the regulating and control subsystem and the fluid dispensing subsystem as shown in Figure 126. All system lines and components containing or exposed to the PGME/water CA liquid are outside the pressure vessel and located in a well-ventilated area.

The fluid supply subsystem consists of the eight components illustrated in Figure 126, plumbed together to provide pressurized liquid to the McDonnell Douglas CA system. Nitrogen for tank pressurization is supplied to the nitrogen pressure regulator at approximately 2,413 KP (350 psig). The regulator is manually adjusted on the ground to reduce the supply pressure to 2,068 +104.4/-0 KPa (300 ±15/-0 psig). The government furnished supply tank has a single port at one end of the tank and a connector containing two ports at the other end. The McDonnell Douglas tank is installed similarly to the Lockheed tank, i.e., single port connector at bottom (liquid outlet) of tank. The pressure regulator is connected to the smallest port of the two-port connector through the pressurization shutoff valve. A servicing overflow line is connected to the remaining port on the two-port connector. This line contains the overflow shutoff valve and provides for draining liquid clear of the aircraft during tank servicing. A connector located between the pressurization shutoff valve and the supply tank provides for attachment of a pressurization vent line. This line contains the pressurization relief valve and the vent valve connected in parallel. This line connects to a scarfed overboard vent mast located to prevent any liquid from

reaching any aircraft surface. The remaining port on the supply tank is the fluid outlet and connects through a filter to the McDonnell Douglas regulating and control subsystem. The filter is accessible for ground cleaning. A tank fill and drain line containing a manual shutoff valve connects through a tee connector to the line from the supply tank to the filter. The tank and all fittings and associated components are suitable for use with a solution of PGME and water.

The nitrogen pressurization system, described in Section 5.6, provides nitrogen flow for purging the fluid dispensing subsystem following spray nozzle operation. This flow is routed through a pressure regulator and the nitrogen purge shutoff valve as shown in Figure 126 to interface with the McDonnell Douglas system. The line connecting the pressure regulator and purge shutoff valve connects through a tee to a pressure relief valve.

The McDonnell Douglas system is described and specified in the appropriate McDonnell Douglas document. This system includes the regulating and control subsystem and the fluid dispensing subsystem. The manual shutoff valve and manual pressure regulator of the regulating and control subsystem are accessible on the ground and provision is made for routing flow overboard during ground adjustment of the regulator.

Operation of the McDonnell Douglas system is from the McDonnell Douglas console. An electrical interlock is provided within the CA control system to ensure that the CA liquid will not flow unless the McDonnell Douglas LETA shield is in the fully deployed position. A limit switch located in the McDonnell Douglas LETA prevents power being applied to open the solenoid shutoff valve of the McDonnell Douglas CA system unless the LETA shield is fully extended.

5.4.3 McDonnell Douglas Shield Ice Protection System

The McDonnell Douglas shield ice protection (IP) system prevents ice accretion by secreting a glycol based fluid through two porous leading-edge panels located on the LETA shield. During aircraft operation in icing conditions with the shield extended, operation of the McDonnell Douglas IP system will prevent ice accretion on the shield if the system is activated shortly after encountering icing conditions.

The McDonnell Douglas IP System is a self-contained system; it does not interface with any other aircraft system except for the electrical interlock described below. The complete system consists of the liquid supply and control subsystem which supplies fluid to the liquid dispensing and metering subsystem as illustrated in Figure 127. The system is controlled through a preassembled control panel located on the McDonnell Douglas control console. All components of the IP system are furnished by McDonnell Douglas including all fittings, connectors, and lines.

The complete IP system consists of the six basic components shown in Figure 127. All components of the IP liquid supply and control subsystem are located inside the pressure vessel. The glycol based IP fluid (Aeroshell 07 or equivalent) is considered to be non-toxic. Access is provided for manual flow rate adjustment of the constant displacement pump. The pump has a nominal flow rate of 3.79 lph (1.0 gph).

An electrical interlock is provided for the IP system to prevent the flow of the IP liquid unless the LETA shield is in the fully deployed position. In addition, a control switch is provided in the cockpit to allow the pilot to activate the IP system independently of the console operator. The limit switch employed is the same switch utilized for the CA system interlock and described in the previous Section. It prevents power from being applied to the IP system pump unit unless the LETA shield is fully extended.

5.5 PURGE SYSTEMS

The possibility of the accumulation of cleaning/anti-icing liquid in the suction system requires that provision be made to remove liquid from these systems prior to initiation of suction. The use of some slots in the Lockheed system for both suction and cleaning also necessitates that the cleaning/anti-icing liquid be removed from these slots and ducting prior to the initiation of suction. The primary purge system provides for purge flow availability above 3,658 m (12,000 ft) altitude. The system uses air from the emergency pressurization system heat exchanger and cannot be used at lower altitudes because of excessive purge air temperatures and excessive temperatures in the aft compartment. This system provides all purge air for the Lockheed and McDonnell Douglas systems at altitudes above 3,658 m (12,000 ft). Operationally, there will be some restrictions that prevent simultaneous purge of both systems under some conditions. A secondary purge system is provided for use by McDonnell Douglas at altitudes up to 3,658 m (12,000 ft). This system uses a portion of the air from the aircraft air conditioning system and interfaces with the primary purge supply line.

The Lockheed air purge system has been designed to remove liquid from the cleaning/anti-icing ducting and to clear all slots of liquid before the initiation of suction. This is necessary to prevent contamination of the suction system with residual cleaning liquid. Due to the numerous points in the system where liquid may be entrapped, the system is designed so the cleaning/anti-icing system may be vented to draw as much liquid as possible back into the tank and then purge the residual liquid out through the slots at altitudes above 3,658 m (12,000 feet). This system interfaces with the suction and cleaning/anti-ice systems.

The McDonnell Douglas clearing system uses purge air from two sources to free the LETA suction system of the contamination avoidance liquid sprayed on the surface during CA operation. The clearing-air requirements have been defined by McDonnell Douglas. For operation below 3,658 m (12,000 feet) altitude, clearing-air will be supplied, as available, by the aircraft air-conditioning system. Above 3,658 m (12,000 feet), the system uses the same air supply and a portion of the same system used by the Lockheed purge system. The system interfaces with the McDonnell Douglas suction system. In addition, the McDonnell Douglas system uses nitrogen to purge the liquid from the CA liquid spray system and nozzles.

5.5.1 Primary Purge System

This system, Figure 128, uses the addition of a valve to an existing aircraft line, modification of the electrical circuitry, and the addition of a line to route air from the existing emergency pressurization line to the 12.7 cm

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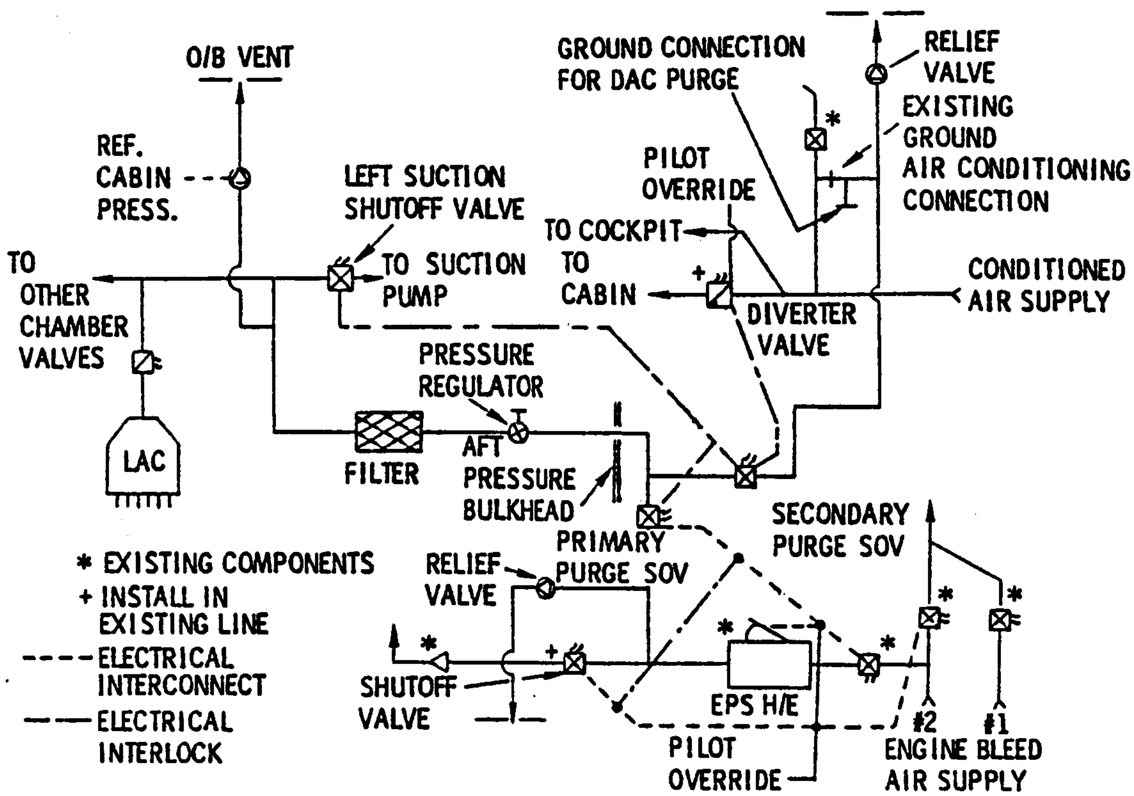

Figure 128. Purge System Schematic

(5.0 in) OD pump interconnection line. The added shutoff (diverter) valve is installed in the discharge line downstream of the emergency pressurization (EP) heat exchanger. The added line tees into the EP heat exchanger discharge line immediately downstream of the heat exchanger, and connects to the primary purge shutoff valve. A pressure relief valve is installed just upstream of the shutoff valve to vent excess pressure. Another line routes the purge flow from the shutoff valve through the aft pressure bulkhead to the vicinity of the chamber valves. Here the flow enters a pressure regulator that is accessible in the cabin. The purge flow is routed from the pressure regulator through a filter to the 12.7 cm (5.0 in) OD duct, at a point between the LEFT suction shutoff valve and the connections to the individual chamber valves. The filter is conveniently located for frequent servicing and for draining of trapped water. The filter consists of a replaceable filter element housed in a specially constructed pressure vessel. A line containing a relief valve tees into the purge supply line downstream of the filter. The relief valve vents excess pressure overboard.

All remotely operated valves in this system are controlled from the aircraft system's control console available to both/either operator. Electrical control interconnections are made to prevent any one of the following three valves from being energized to the open position unless the remaining two valves are fully closed:

(1) LEFT Suction Shutoff Valve

(2) Primary Purge Shutoff Valve

(3) Secondary Purge Shutoff Valve

Actuation of the primary purge system from the consolette causes the left-hand air conditioning refrigerator assembly to be shut down and the EP ram air control scoop to move to the full-open position. These events normally occur when emergency pressurization is selected. A separate switch is provided on the consolette to adjust scoop position from the full open position and thus control purge air temperature. The electrical system provides for selection of "emergency cabin air pressurization" to override the selection of the purge system and restore the aircraft system to its normal emergency operation. Selection of "emergency cabin air pressurization" also prevents actuation of the purge system.

5.5.2 Secondary Purge System

This system also includes the addition of a diverter type control valve to an existing aircraft duct and an additional line to route cabin air flow to the primary purge system line as illustrated in Figure 128. The added diverter valve is installed in the cabin air duct immediately downstream of the point where flow from the refrigerator bypass valve enters the duct. A new duct is installed between the existing ground air conditioning connection and the primary purge system line at a point downstream of the primary purge shutoff valve and upstream of the purge pressure regulator. This line contains a new ground connection interface followed downstream by a remotely-controlled shutoff valve. The new ground connection point is provided for purging the McDonnell Douglas LETA from a ground cart following flight tests. A line containing a relief valve tees into the line between the existing ground air-conditioning connection and the shutoff valve to vent excess pressure to protect the air-conditioning ducts.

The secondary purge shutoff valve is electrically interlocked with the primary purge shutoff valve and the LEFT suction shutoff valve to protect new and existing ducts and components from excessive pressure that could result from improper purge system operation. Another electrical interlock prevents the secondary purge system shutoff valve from being in a closed position simultaneously with the cabin air diverter valve.

5.6 NITROGEN PRESSURIZATION SYSTEM

A high pressure gaseous nitrogen supply is used to provide pressurization for the liquid reservoirs of both the Lockheed and McDonnell Douglas liquid systems, McDonnell Douglas contamination avoidance (CA) system purge, and instrumentation purge. The system also provides pneumatic operation of the purge system shutoff valves and pressure regulator. The nitrogen source provides nitrogen at a regulated pressure of about 2,413 KPa (350 psig). The system configuration is shown schematically in Figure 129. The nitrogen is routed from the pressure vessel through a pressure regulator to a shutoff valve. For tank servicing, a line containing a manual shutoff valve is teed into the line immediately downstream of the pressure vessel. Dual pressure relief valves, arranged in parallel, are used between the tank outlet and the pressure regulator to vent excess tank pressure overboard. The flow is routed from the

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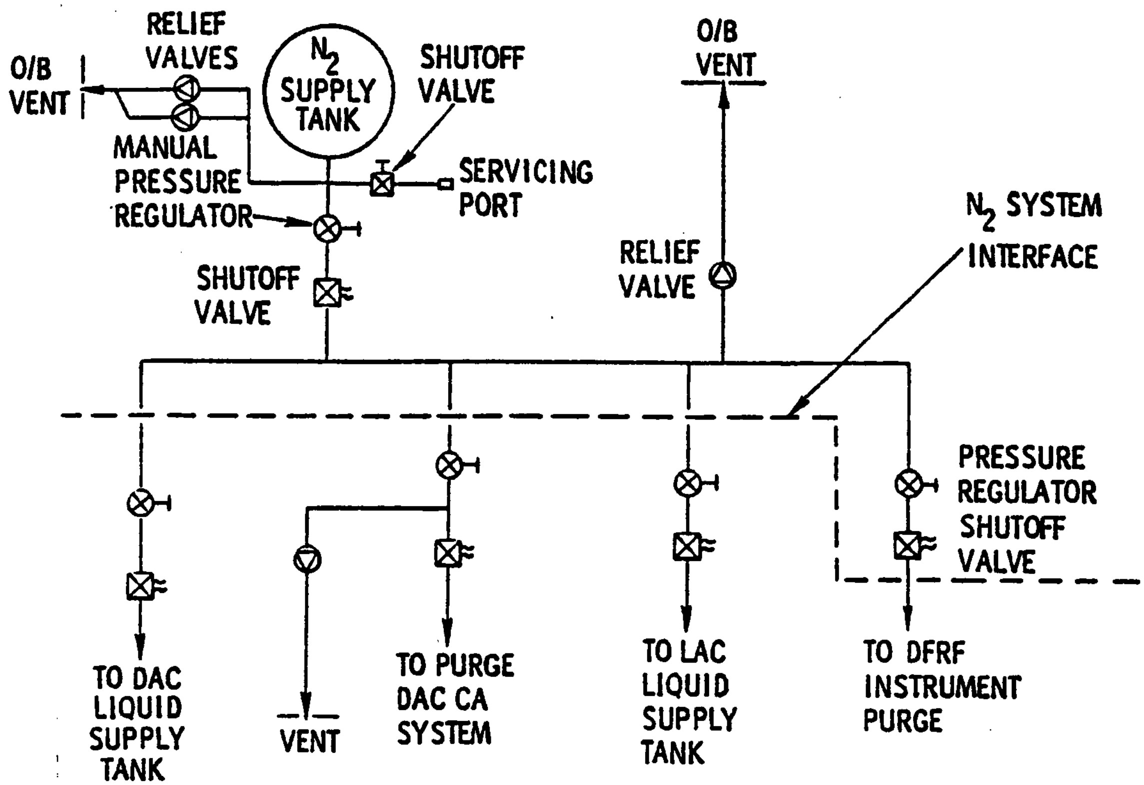

Figure 129. Nitrogen Pressurization System

shutoff valve to the McDonnell Douglas CA system nitrogen pressure regulator, McDonnell Douglas CA system nitrogen purge pressure regulator, the Lockheed cleaning/anti-icing system nitrogen pressure regulator, and the purge system shutoff valves. In addition, flow is provided to the instrument purge pressure regulator and shutoff valve located in the vicinity of the nitrogen supply tank. The plumbing downstream of the instrument purge shutoff valve is the responsibility of the Dryden Flight Research Facility (DFRF). To provide protection from excessive regulated pressure a line is teed into the nitrogen line downstream of the pressurization shutoff valve and connnects to an overboard vent through a pressure relief valve.

5.7 INSTRUMENTATION, DISPLAYS, AND CONSOLES

This section contains a description of the Lockheed leading-edge test article instrumentation, the recommended aircraft systems instrumentation for the JetStar leading-edge glove flight test program and data display recommendations. The Lockheed console and the aircraft LFC systems controls are also included. All of the above will be provided by the DFRF except as noted herein. The aircraft systems instrumentation includes only that portion of the system up to the interfaces with the McDonnell Douglas designed systems. The LFC systems controls are described up to their interface with the dedicated Lockheed and McDonnell Douglas systems controls. For clarity, these systems are grouped by function such that each description includes both the Lockheed and the aircraft common systems. All data are to be recorded on the DFRF furnished PCM system unless otherwise noted.

The material contained herein is believed to be consistent with the chamber valve assembly, to be separately provided by NASA LaRC for control of the slot suction flows via remote valves, the NASA DFRF provided onboard computer capability, and NASA/DFRF provided consoles including a CRT and keyboard. The aircraft common systems are consistent with the Lockheed understanding of McDonnell Douglas requirements. The data displays currently planned for the flight test program have been developed by NASA DFRF and NASA Langley. These displays vary somewhat from the Lockheed recommended displays shown in this section but substantially incorporate all the recommended information for display with a revised parameter code and some additional parameters.

The instrumentation recommendations of this section are based upon a limited number of available data points. This limitation is imposed by the number of available multi-port pressure measuring devices and the data recording system's PCM capability. Thus, the recommendations given here reflect a judgment on the best use of a limited number of data points and not an instrumentation system to completely define system performance in operation.

5.7.1 General Instrumentation Recommendations

The description of each functional system includes a discussion of instrumentation recommendations, a schematic diagram of the complete system, a list of the instrumentation requirements, and a description of the required CRT data displays. Numbers are included on the schematic to identify instrumentation items and provide a key for correlating instrumentation on the schematics, instrumentation lists, and consoles. The instrument lists are tables containing the pertinent requirements for each instrument, such as number required and ranges. For simplicity, only example slots are shown on the schematics, but the appropriate number of slots is stated.

5.7.2 General Display Recommendations

Data displays fall into three general categories:

(1) Direct-reading continuous displays for system adjustment and monitoring. These may be remotely indicated, or in some cases, located at a point where they may be observed while a setting, such as a pressure regulator, is being adjusted. These may or may not be duplicated on the CRT and do not require computer calculations.

(2) Tabular digital displays for reference or intermittent monitoring or adjustment. These are tabular items displayed on the CRT that are not amenable to an analog plot but are required intermittently.

(3) Analog displays for performance assessment and adjustment. These are obviously on the CRT and may or may not require computer evaluation.

These displays are of direct reading or on-line computer output data with analysis and/or stored data as necessary. These displays may include digital or analog test data as well as stored reference data for comparisons. It is recommended that the CRT have the capability for separate simultaneous display of two independent data sets or for expanding a single analog display to fill the entire screen. As an example, the operator may desire a display of spanwise

distributed suction versus wing station with a simultaneous, separate display of needle valve position versus slot number. Alternatively, the operator might desire a single display of slot suction flow versus slot number that fills the CRT screen for better resolution.

Simultaneous display of any two independent data sets should be at the option of the operator and controlled from the keyboard. In addition to a display of analog or digital test data, the operator should have the option of an overlay display of stored data for some functions to assist in evaluation of test conditions or adjustment of controls such as a display of desired needle valve positions superimposed on a display of actual needle valve position versus slot number. Analog displays should include suitably identified grid scales and overlayed displays should be identified so the operator can readily distinguish test data from stored data overlays. Each display should be suitably identified so the operator may readily identify simultaneous independent displays. Displays illustrated in this section show test data as a solid line and stored data as a dashed line.

5.7.3 Basic Aircraft Parameters

It is assumed that the existing aircraft test instrumentation for basic flight parameters will be adequate for the subject test program and will be available to the console operator. Therefore, these data are not included on the Lockheed console nor are they discussed here. These parameters include:

Parameter	Units
(1) Pressure altitude	Feet
(2) Flight Mach number	Number
(3) True air speed	KTAS
(4) Ram air temperature	$^{\circ}R$
(5) Ambient temperature	$^{\circ}R$
(6) Ambient static pressure	psia
(7) Free-stream total pressure	psia
(8) Angle of attack	degrees
(9) Angle of yaw	degrees -right/left
(10) Cabin pressure	psia (or psid rel. to amb.)
(11) Time of day	Hrs/Min
(12) Aircraft weight (keyboard entry)	lbs

In addition to the above, NASA will provide measurement of the aircraft electrical charge and ice-particle measurement with a Knollenberg probe. The first four of the above, along with items (6), (8), and (10), should be continuously displayed in the cabin on dedicated digital instruments for the use of both console operators. Items (1) through (11) should be recorded on the PCM for all test points.

It is recommended that the data shown in Figure 130 (Display 1) be available as a call-up. All parameter names may be abbreviated or coded and a display of units is not necessary as a continuous display, thus requiring only two lines on the CRT. In addition, the display includes a line of codes defining the test plan conditions and a line displaying the desired altitude, Mach number, and angle of attack associated with the desired test conditions.

Test Point XXX   Test Cond. - (Code)   Mode XXXX   Rec. No. XXX

	Alt.	M_n	Tot. Temp.	Tot. Pr.	Amb. Temp.	Amb. Pr.	Alpha	Gamma	Cabin Pr.
	Ft.	No.	°R	PSIA	°R	PSIA	°Ang.	°Ang.	PSIA (PSID)
ACT	XXXXX	.XXX	XXX	XXXX	XXX	XXXX	XX.X	X.XX	XXXX
DES	XXXXX	.XXX					XX.X		

Figure 130.   Basic Aircraft CRT-Display 1

5.7.4 Suction System

The suction system instrumentation has been selected to meet the requirements for this program within the constraints of the limited number of data points and to provide information to the console operator to permit in-flight system adjustments. The instrumentation for the Lockheed suction system provides for measurement of the total suction flow through each slot. Within the limits of available pressure instrumentation, additional instrumentation is included to permit evaluation of the spanwise slot flow distribution when analyzed in conjunction with airfoil surface C_p data. These data will permit limited analytical evaluation of the slot and internal ducting system characteristics.

Instrumentation is also outlined for the aircraft suction pump system that provides the suction source common to both the McDonnell Douglas and Lockheed systems. This instrumentation provides data for the operation and control of the suction pump. This system was shown in Figure 121. Pertinent requirements for this instrumentation are presented in Table 9.

5.7.4.1 Lockheed Leading-Edge Suction System Instrumentation

The Lockheed LETA and suction flow and control instrumentation is shown schematically in Figure 131 and Table 9. The McDonnell Douglas system is also shown for clarity, but definition of this instrumentation will be provided by McDonnell Douglas. Four types of slots are shown in the schematic diagram for the Lockheed LETA; these include cleaning (C), dual purpose (D), lower (L) and upper (U) surface slots. Due to the intermittent operation of the C and D slots with liquid, no internal pressure instrumentation is included in these slots for suction data, although the suction flows of both are measured. The U and L slots are dedicated suction slots in the upper and lower surfaces, respectively. All suction system instrumentation located within the LETA is in the form of pressure taps located in either U or L slots, which are routed to the inboard end of the LETA interface as provided by Lockheed.

Instrumentation taps indicated as (2) in Figure 131 and in Table 9 are located within the internal slot ducting of selected U and L slots. These taps are located in slots U1, U2, U4, U6, U8, L1, L3, and L5 and are located at both the inboard and outboard ends of the ducting. They consist of 0.102 cm (0.040 in) OD 0.07 cm (0.028 ID) stainless-steel tubes with the inner end located approximately flush with the inner wall of the collector duct at least 2.54 cm

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TABLE 9. LOCKHEED LEADING EDGE SUCTION SYSTEM INSTALLATION

KEY NUMBER ON FIGURE 131	PARAMETER	NUMBER OF ITEMS	UNITS	RANGE	ACCURACY
(1)-(2)	PRESSURE DROP FROM WING SURFACE TO SLOT COLLECTOR DUCT (2/SLOT)	16	PSID	0 TO 1.0	±0.5%
(2A)	LETA SLOT LINE INTERFACE PRESS. (SLOTS U2, U4, U6 AND L5)	4	PSID*	0 TO 2.5	±1.0%
(3)	CHAMBER VALVE TEMPERATURE	2	°F	-50 TO 180	±2.0%
(4)	NEEDLE VALVE PRESSURES	26	(LARC)	(LARC)	(LARC)
(5)	CHAMBER VALVE PRESSURE	2	PSID**	0 TO 15	± 1.0%
(6)	SLOT NEEDLE VALVE POSITION	26	IN	0 TO 2.0	±0.001 IN
(7)	SUCTION CHAMBER PRESSURE CONTROL VALVE POSITION	2	DEG. ANGLE	0 TO 90	±1 DEG
	NEEDLE VALVE POTENTIOMETER SETTING (NOTE: THIS DATA OBTAINED FROM CONSOLE BUT NOT AUTOMATICALLY RECORDED ON PCM)	26	NUMBER	0 TO 999	—

* RELATIVE TO AMBIENT

** RELATIVE TO CABIN PRESSURE

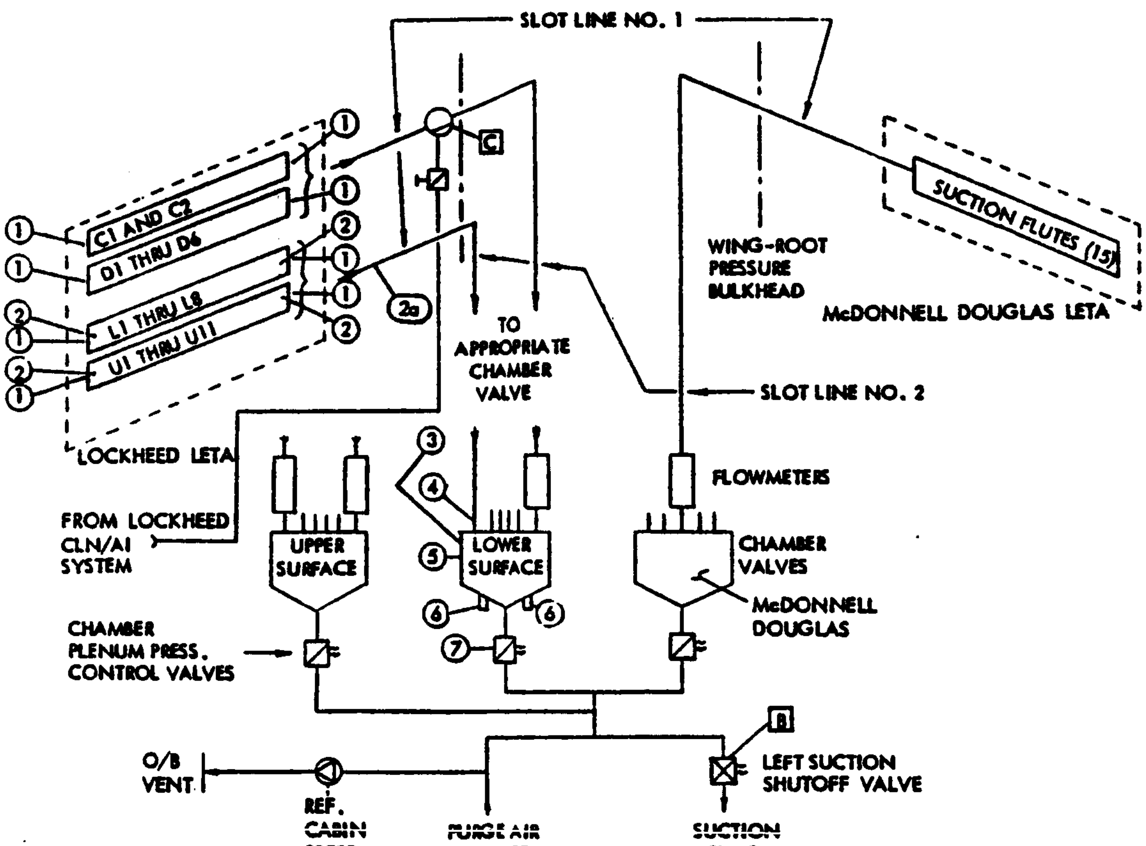

Figure 131. Leading Edge Suction System Schematic

(1 in) beyond the blocked end of the slot/slot duct. This instrumentation will be used to evaluate the spanwise flow variation along the slot and the significance of the data lies in the very low pressure differentials relative to the local airfoil surface pressures. Therefore, it is necessary to measure these pressures directly as individual differentials, ΔP (1) - (2), at each end of the affected slots relative to the local surface static pressure, the taps for which are located just inboard and outboard of the ends of the active suction slots as shown by (1) in Figures 131 and 142 and Table 14. These 16 surface static taps (1) also form a part of the surface instrumentation described in Sections 4.1.3 and 5.7.8 hence must be "teed" to separate pressure sensors for measurement of the local surface pressure. The instrumentation taps at both (1) and (2) must be purged any time the cleaning liquid system is turned on.

Due to the small ΔP range of the transducers necessary for accurate measurements, it may be necessary to protect these instruments from damage during sea-level checkout and during system purge. DFRF has proposed a means by which these transducers may be isolated during these operations. For this reason, the suction system instrumentation list of Table 9 includes only cruise ranges for these instruments. The accuracies specified are related to the maximum operating range values for cruise conditions.

Four slot line suction pressures will be measured at the LETA interface as indicated by (2a) in Figure 131 and Table 9. Pressure taps will be located in the lines of slots U2, U4, U6, and L5. These four slots were selected from among the eight slots having internal pressure taps to provide a more detailed breakdown on the ducting system performance in the LETA.

Individual slot flow total pressures will be measured at the needle valve inlets of all slots as indicated by (4) on the schematic and table. It should be noted that slots C1 and C2 have a common piccolo tube inside the LETA so that only a single combined flow total pressure is required for these two slots. The pressure probes for these measurements are to be incorporated in the needle valves as provided by NASA LaRC. Slot flow measurement capability will be provided by NASA LaRC through calibration of the sonic needle valves.

The chamber valves are also to be provided by NASA LaRC. Provision will be included for measurement of the chamber valve plenum pressure (5) and temperature (3) as indicated on the schematic and table. The two chamber valves in the Lockheed system are nominally designated for the upper and lower LETA surface suctions. These chamber valves include 13 sonic needle valves each for individual control of the slot flows. NASA LaRC will provide controls and position sensors (6) for each of these needle valves. The sonic needle valves will be calibrated by NASA LaRC for on-line calculation and CRT display of individual slot/flute mass flows. This calibration is to be based upon the instrumentation identified in this section.

A motorized, controllable valve is located downstream of each chamber valve, including the McDonnell Douglas system. In the suction mode, it is expected these valves will be in the full-open or full-closed position, depending on whether suction is being applied to the associated chamber valve system. However, in the purge mode described later, these valves will normally be set to an intermediate position. Therefore, a position indicator is provided for these valves as indicated by (7) in the schematic and table.

5.7.4.2 Suction Pump Instrumentation

Figure 121 provides a schematic of the suction pump system. The pump instrumentation is listed in Table 10. Sensors for (11), (12), (14), (15), and (16), at a minimum, will be provided by AiResearch.

5.7.4.3 Suction System Display Recommendations

The suction system displays outlined herein have been recommended to provide the greatest flexibility to the operator for adjusting the systems and interpreting the flight data. They are considered to be the minimum slot suction system requirements for an effective flight test program. The pump data requirements are representative and further changes may be anticipated when the pump modifications are fully defined.

5.7.4.4 Suction System Displays

Displays (2) and (3), Figures 132 and 133 are provided to the console and consolette operators for setting of the suction system and for monitoring the suction pump operation. They may also be used in setting the system for purge operation. It is recommended that the single line display of pump data be provided on the comparable McDonnell Douglas display to be used when the operator of the McDonnell Douglas system is also the consolette operator. Upper- and lower-surface data are displayed individually.

TABLE 10. SUCTION PUMP CHARACTERISTICS

KEY NUMBER ON FIGURE 121	PARAMETER	ITEMS	UNITS	RANGE	ACCURACY
(8)	TOTAL LFC SUCTION FLOW PRESSURE	1	PSID*	0 TO 3.0	±1.0%
(9)	TOTAL LFC SUCTION FLOW TEMPERATURE	1	°F	-65 TO 180	±2.0°
(10)	PUMP INLET PRESSURE	1	PSID*	0 TO 3.0	±1.0%
(11)	PUMP INLET TEMPERATURE	1	°F	-65 TO 180	±2.0°
(12)	PUMP DISCHARGE PRESSURE	1	PSID*	0 TO 1.0	±1.0%
(13)	PUMP DISCHARGE TEMPERATURE	1	°F	0 TO 600	±2.0°
(14)	PUMP ROTOR SPEED - PCM AND DIG. DISP. ON PUMP CONTROL PANEL	1	RPM	0 TO 60,000	±100 RPM
(15)	OVERSPEED WARNING LIGHT ON PUMP CONTROL PANEL	1	-	ON-OFF	-
(16)	OIL PRESSURE WARNING LIGHT ON PUMP CONTROL PANEL	1	-	ON-OFF	-
(17)	OIL IN TEMPERATURE DIAL GAUGE LOCKHEED CONSOLE	1	°F	-65 TO 300	±2°
(18)	TURBINE INLET PRESSURE	1	PSID*	0 TO 150	±1.0%

*RELATIVE TO AMBIENT

NOTE: ITEMS (10) THROUGH (16) ARE TO BE PROVIDED BY AIRESEARCH

1. CHAM VALVE UPPER SURFACE DES POS U XXX.X L XXX.X D XXX.X

2. SET CODE XXXXX ACT POS XXX.X XXX.X XXX.X SUCT FLO PR XX.XXX
(7) (7) (7) (8)

3. PURGE LINE PR XX.XX DP CAB PR XX.XX XX.XX XX.XX
(41) (5) (5) (5)

4. PURGE LINE PR XX.XX DP AMB TEMP XXX.X XXX.X XXX.X
(41) (3) (3) (3)

5. PUMP DATA DR PR XXX.X IN PR XX.XX INT XXX DIS PR XX.XX N XXXXX RC X.XX
(18) (11) (12) (13) (14)

6. SLOT C D1 U1 U2 U3 U4 U5 U6 U7 U8 U9 U10 U11

7. DP 12 X.XXX X.XXX X.XXX X.XXX X.XXX
(1)-(2) (1)-(2) (1)-(2) (1)-(2) (1)-(2)

8. POTS XXX XXX XXX XXX XXX XXX XXX XXX XXX XXX XXX XXX XXX

9. POSS X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX

10. POSA X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX
(6) (6) (6) (6) (6) (6) (6) (6) (6) (6) (6) (6) (6)

11. PR X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX
(4) (4) (4) (4) (4) (4) (4) (4) (4) (4) (4) (4) (4)

12.

Figure 132. Basic Aircraft CRT-Display 2

13. CHAM VALVE LOWER SURFACE DES POS U XXX.X L XXX.X D XXX.X

14. SET CODE XXXXX ACT POS XXX.X XXX.X XXX.X SUCT FLO PR XX.XXX
(7) (7) (7) (8)

15. PURGE LINE PR XX.XX DP CAB PR XX.XX XX.XX XX.XX
(41) (5) (5) (5)

16. PURGE LINE PR XX.XX DP AMB TEMP XXX.X XXX.X XXX.X
(41) (3) (3) (3)

17. PUMP DATA DP PR XXX.X IN PR XX.XX INT XXX DIS PR XX.XX N XXXXX RC X.XX

18. SLOT D2 D3 D4 D5 D6 L1 L2 L3 L4 L5 L6 L7 L8

19. DP 12 X.XXX X.XXX X.XXX
(1)-(2) (1)-(2) (1)-(2)

20. POTS XXX XXX XXX XXX XXX XXX XXX XXX XXX XXX XXX XXX XXX

21. POSS X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX

22. POSA X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX X.XXX
(6) (6) (6) (6) (6) (6) (6) (6) (6) (6) (6) (6) (6)

23. PR X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX X.XX
(4) (4) (4) (4) (4) (4) (4) (4) (4) (4) (4) (4) (4)

Figure 133. Basic Aircraft CRT-Display 3

The upper portion of this display is pertinent to the basic suction system and provides data common to all slots by displaying chamber valve data. Purge data are also included so this same display may be used when setting the needle valves for the purge modes. The purge line pressures are measured as P relative to the cabin and computed values are also shown of purge line pressure in P relative to ambient. These pressures should be used only for monitoring purposes. The pump data includes a computed value for pump pressure ratio (Rc) for monitoring compressor performance, along with other pump measured parameters.

The lower portion of the displays is associated with individual slot needle valves designated by the line entitled SLOT. These data include the slot collector duct (2) to surface pressure (1) differentials for the eight slots (five upper surface and three lower surface) instrumented for these data, labeled DP12 on the display. The next line displays the desired needle valve potentiometer settings labeled POTS. This display is from pre-flight or previously stored data identified by the SET CODE in the upper portion of the display. It is recommended that the operator have the capability to enter a SET CODE that has not previously been identified, and then type in potentiometer settings that the operator has established from flight adjustments and observations so they may be recorded or stored for duplication later. The next line labeled POSA includes desired needle valve position indications and is similarly displayed from pre-flight or previously stored data identified by the SET CODE. The line of data labeled POSS is actual measured needle valve positions. The operator should have the capability of storing these under a new SET CODE as POSA data for future use. The last line, PR, is the pressures measured at the entrance to each needle valve as being provided by the NASA LaRC.

5.7.4.5 Leading-Edge Test Article Displays

The following analog displays should be available to the operator to assess system performance and to make adjustments to the system. They should be available as single displays that fill the CRT screen or as paired displays with a simultaneous, separate display of any other data set display identified in this section, each displayed on a separate portion of the screen. Desired values should also be stored for overlay display as indicated and should be stored so they may be readily changed as the test progresses. It is desirable that multiple stored sets of desired values be available for each data set at any time.

Display 4, Figure 134, Total Distributed Suction Flow

Display 4 presents the total slot suction flow data as if distributed uniformly across the span of the test article and is useful for overall adjustment of chordwise distribution and comparison to the desired value. These flow data are evaluated from needle valve pressures, positions and chamber valve temperature. The desired values are available from stored data for the condition code and from flight ambient pressure and temperature, and flight Mach number.

Display 5, Figure 135, Slot Spanwise Flow Distribution

Eight slots (U1, U2, U4, U6, U8, L1, L3, and L5) are instrumented for internal duct pressure taps, (2), at both ends. These data, together with external surface static taps, (1), may be used to determine the spanwise distri-

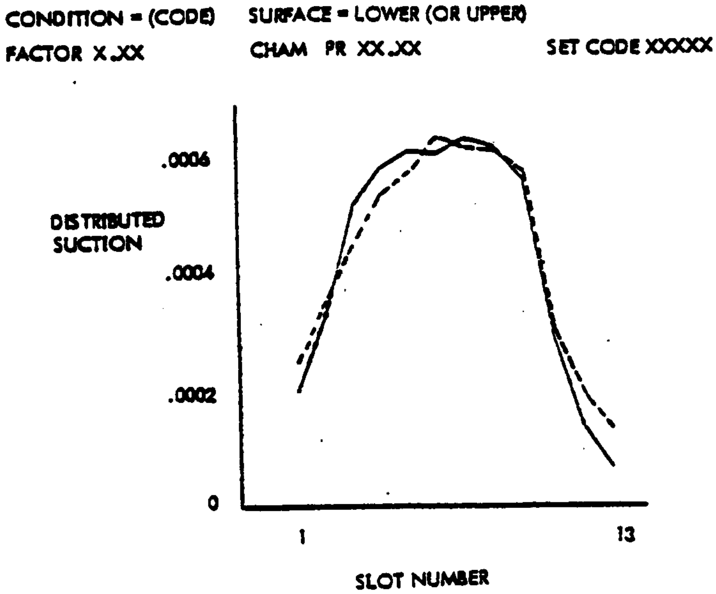

Figure 134. Total Distributed Suction Flow-Display 4

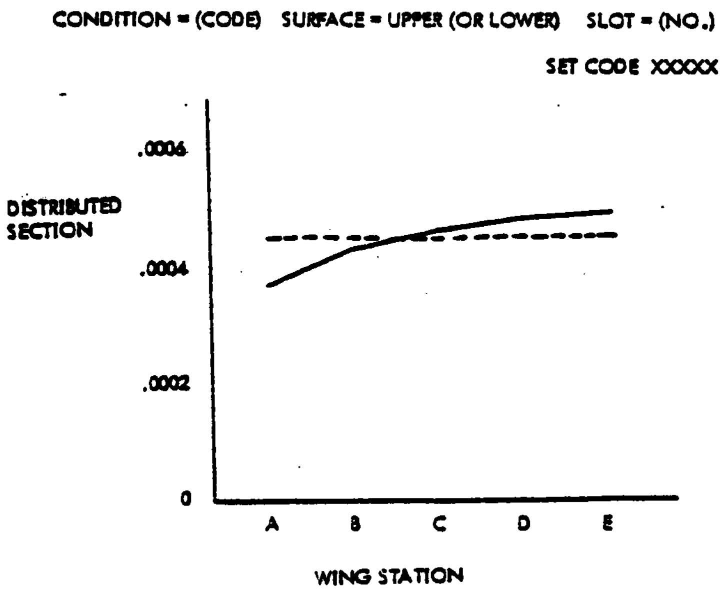

Figure 135. Slot Spanwise Flow Distribution-Display 5

bution of slot suction flow for these eight slots as shown in Display 5. These data provide an insight into the local deviations in spanwise slot flow as a result of spanwise surface pressure gradients. This will allow the operator to adjust the distributed suction flows to assure meeting a minimum desired flow over any segment of these slots, and together with Display 4, will provide similar guidance for all slots. In addition to the data requirements noted, this display requires the same data as Display 4.

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Display 6, Figure 136, Local Distributed Suction Flow

Utilizing the same data as Display 5, the data for the eight instrumented slots of Display 5 may be displayed as a crossplot in the same format as Display 4 providing a limited chordwise suction distribution at each of five spanwise locations corresponding to the spanwise locations of the chordwise rows of the surface static taps. These plots, as shown in Display 6, will provide an indication of any cumulative under- or over-suction characteristic that may result from spanwise surface pressure distribution. It is hoped that a means of satisfactorily correlating these data with the surface pressure distributions at the noninstrumented slots may provide a more complete display of local chordwise suction flow distribution, but the display of the instrumented slot data is quite valuable without this refinement.

5.7.4.6 Aircraft LFC System's Displays

All instrumentation requiring continuous display for pump control and monitoring will be mounted on the McDonnell Douglas console for the use of the designated aircraft systems operator. An exception is the pump oil temperature dial gauge to be installed near the top of the Lockheed console. This installation will allow the gauge to be viewed from either console position. A master pump arming switch will be located in the cockpit where the pilot will have ultimate control over the pump. AiResearch is to provide a pump control panel for installation in the McDonnell Douglas console for control and troubleshooting of the TSP system. This panel includes pump start and start switches and a continuous display of rotational speed, Item 14 in Figure 121. Also included are (1) indicator lights for the pump master switch, pump overspeed trips, and low oil pressure warning, (2) an electronic controller circuit, which is a major component of the automatic surge control system, (3) an automatic/manual mode surge control switch, (4) a potentiometer for

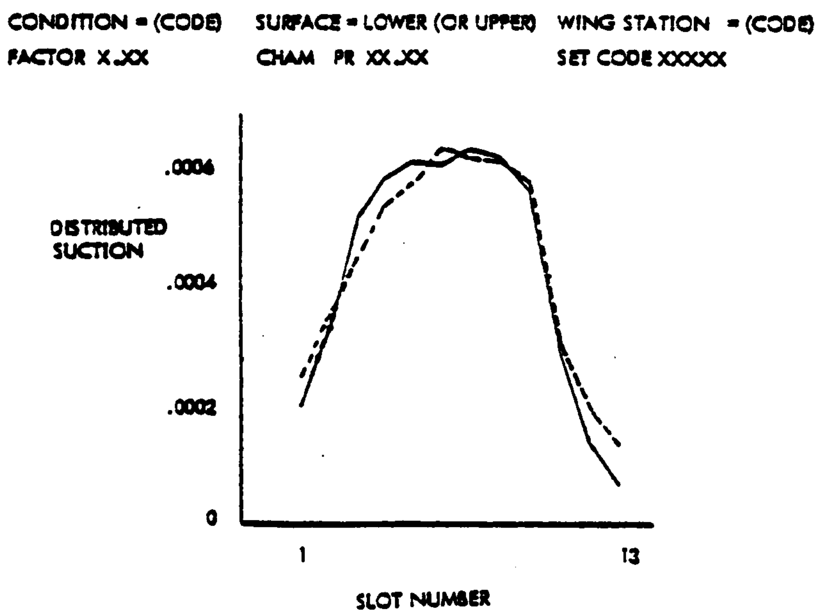

Figure 136. Lockheed Local Distributed Suction Flow-Display 6

positioning the vent valves in the manual mode, and (5) various banana plug receptacles for test use only and for a DC analog output for pump speed. It is presumed in this document that all other instrumentation, items 8, 9, 17, 18 in Figure 121, must be supplied by the NASA DFRF as required. Figure 121 indicates lighted switches (A and B) on the turbine bleed air supply shutoff valve and the suction shutoff valve. The switch for the turbine air supply is included on the AiResearch pump control panel, but a lighted suction shutoff valve switch will be provided by the NASA DFRF and mounted on the McDonnell Douglas console. The pilot's panel will also include a NASA DFRF furnished turbine bleed air supply arming switch in series with the pump panel shutoff switch. The pump overspeed warning, Item (15) in Figure 121 and the pump oil pressure warning, (16), will be included on the AiResearch provided panel as red lights. Duplication of these warning lights or the pump speed indication on the pilot's panel is at the option of NASA DFRF. The overspeed and oil pressure will be provided on the pump by AiResearch. All other instrumentation items required for monitoring pump operation are included on the CRT digital display as part of Displays 2 and 3 and should also be included on similar McDonnell Douglas CRT displays. When more is known of the pump operation, it may be desirable to include additional displays for monitoring purposes. All items should be recorded by the PCM system.

5.7.5 Cleaning and Contamination Avoidance Systems

Unlike the suction system, there is no commonality between the Lockheed and McDonnell Douglas liquid systems; therefore, they will be discussed separately. The complete Lockheed system is included herein while only the aircraft liquid supply portion of the McDonnell Douglas system will be included up to the point of the interface with the McDonnell Douglas designed system. Identification of all instrumentation beyond the McDonnell Douglas interface is their responsibility. The instrumentation in the Lockheed system provides for safe operation of the system and only limited in-flight adjustment and performance evaluation. The instrumentation in the McDonnell Douglas liquid supply system provides for monitoring the system supply and for ground setting only.

5.7.5.1 Lockheed Cleaning/Anti-Ice System Instrumentation

The Lockheed system is illustrated schematically in Figure 137, and pertinent required instrumentation information is included in Table 11. All instrumentation indicated in this system must be suitable for use with PGME. The liquid temperature sensor indicated at (20) is located immediately outside the leading-edge test article at the inboard location in the duct for slot D1. The pressure at (21) should be located as near as possible to the manifold for the cleaning slot flow distribution in the common line carrying fluid to the manifold or optionally may be located in the manifold. This instrumentation should be close-coupled to the line to minimize any effects of entrapped air or should be primed each time liquid is to be used in the system. The liquid temperature sensor (23) should be located immediately upstream of the total liquid flowmeter (22). The pressure at (24) should be located as near as possible to the liquid tank outlet and installed so as to minimize any effects of entrapped air or primed. All the above instrumentation should be compatible with recording on the PCM data system. The pressure measurement at (25) is made immediately downstream of the nitrogen pressure regulator and should be of a visual

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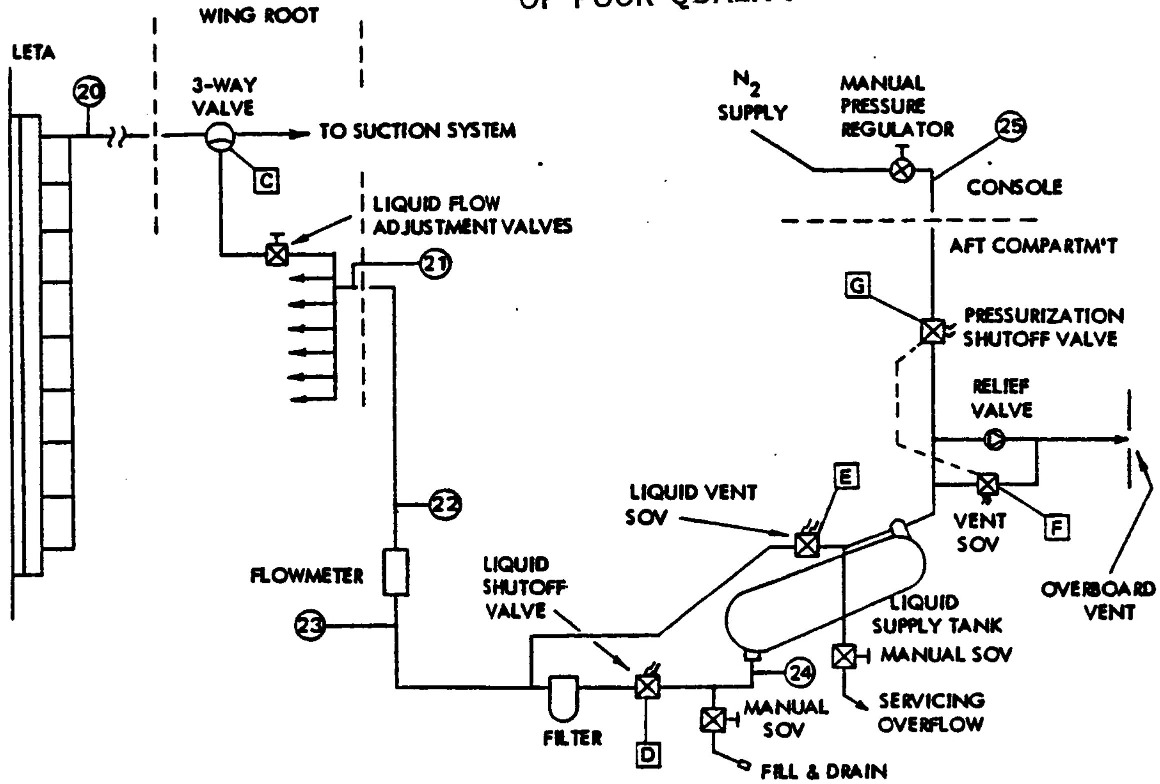

Figure 137. Lockheed Cleaning/Anti-Icing System

TABLE 11. LOCKHEED CLEANING/ANTI-ICING INSTRUMENTATION

KEY NUMBER ON FIGURE 137	PARAMETER	NUMBER OF ITEMS	UNITS	RANGE	ACCURACY
20	SLOT LIQUID TEMPERATURE	1	°F	-40 TO 160	± 2%
21	LIQUID MANIFOLD PRESSURE	1	PSID	0 TO 30	± 1%
22	LIQUID FLOW RATE	1	LB/MIN	0 to 16.0	± 1%
23	LIQUID LINE TEMPERATURE	1	°F	-40 TO 160	± 2%
24	LIQUID TANK PRESSURE	1	PSID	0 TO 30	± 1%
25	LIQUID SYSTEM PRESSURIZATION - (NITROGEN DIAL PRESSURE GAUGE ON CONSOLE)	1	PSID	0 TO 30	± 1%

direct reading dial or digital type read-out. The regulator and pressure read-out are located on the console for in-flight adjustment of the cleaning-liquid supply tank pressure, thereby adjusting cleaning liquid flow, see Figures 141 and 149.

5.7.5.2 McDonnell Douglas Contamination Avoidance Liquid Supply Instrumentation

Since only the Mcdonnell Douglas liquid supply system instrumentation is included in this document, it is very limited as shown in Figure 138 and listed in Table 12. The pressure instruments located at (30) and (31) are both in the nitrogen pressurization system and should be located immediately downstream of their respective pressure regulators for use in ground setting of the regulators, also see Figure 141. They are both of the visual-dial type. The two pressures, (32) and (33), are both located immediately downstream of the PGME liquid filter and must be suitable for use with PGME. The instrument at (32) is a visual-dial indicating type to be used in ground setting of supply tank pressure. The instrumentation at (33) is for remote indication of this same pressure and must be compatible with the PCM data system. Both must be installed in a manner to minimize any effects of entrapped air or must be primed each time the system is to be used.

5.7.5.3 Cleaning And Contamination Avoidance Systems Displays

A recommended display for the Lockheed system is described, while recommendations are made for McDonnell Douglas system displays to be included with other McDonnell Douglas requirements on their console. These display recommendations include both those recommended for the CRT as well as for the console itself. All CRT instrumentation should be recorded on the PCM system.

Lockheed Cleaning/Anti-Ice System Displays

The recommended display for the Lockheed system is based on safe system operation with moderate provision for inflight adjustment. Items (20) through (24) should be displayed digitally on the CRT in a simple tabular form as shown in Display 7, Figure 139. The CRT display should be augmented by a display of the basic aircraft instrumentation, but it may be accomplished by a call-up of Display 1.

Item (25) should be a direct-reading dial pressure gage located on the console near the adjustable nitrogen pressure regulator to be used for in-flight adjustments to the liquid flow rate. In addition, lighted switches should be provided on the console for the system remote shutoff valves indicated by D through G. Valves G and F are connected to a common nitrogen supply switch lighted green when G is closed and F is open and lighted amber for the reverse condition. It is recommended that switches E and D be lighted green for the closed position and amber for the open position. There are seven 3-way valves. These should have a common (single) lighted switch, (C); amber is recommended for open to the cleaning system and green for the position open to the suction sytem.

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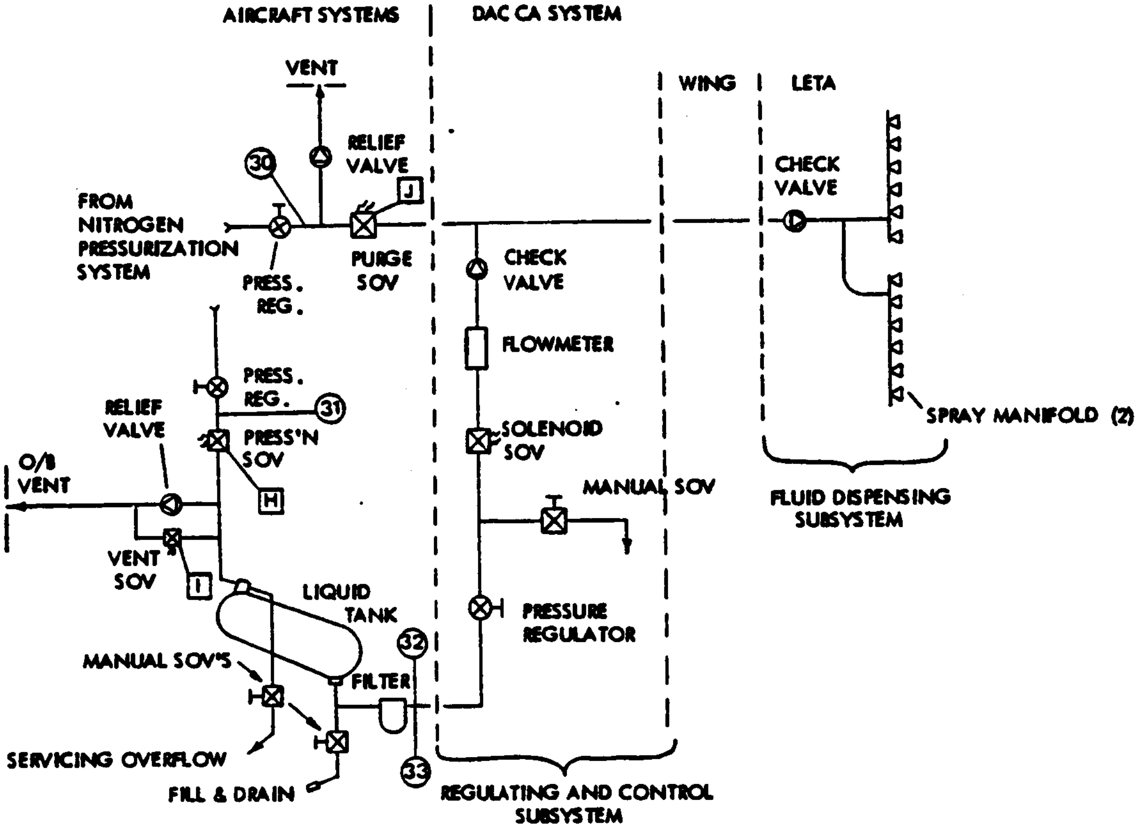

Figure 138. McDonnell Douglas Contamination Avoidance System

TABLE 12. McDONNELL DOUGLAS CONTAMINATION AVOIDANCE LIQUID SUPPLY INSTRUMENTATION

KEY NUMBER ON FIGURE 138	PARAMETER	NUMBER OF ITEMS	UNITS	RANGE	ACCURACY
30	NITROGEN PURGE SUPPLY PRESSURE TO LIQUID LINE (DIAL PRESSURE GAUGE IN AFT COMPARTMENT)	1	PSID	0 - 350	± 1%
31	NITROGEN SUPPLY PRESSURE TO LIQUID TANK (DIAL PRESSURE GAUGE IN AFT COMPARTMENT)	1	PSID	0 - 350	± 1%
32	LIQUID SUPPLY PRESSURE TO DAC SYSTEM (DIAL PRESSURE GAUGE IN AFT COMPARTMENT)	1	PSID	0 - 350	± 1%
33	LIQUID SUPPLY PRESSURE TO DAC SYSTEM (FOR DATA RECORDING AND CRT DISPLAY)	1	PSID	0 - 350	± 1%

McDonnell Douglas Contamination Avoidance Liquid System Displays

The displays recommended for McDonnell Douglas are only provided to supplement their desired data with required data for the liquid supply system and for ground setting of the supply system. Items (30), (31), and (32) in Figure 138 are dial-indicating pressure gages and should be so located that they may be observed while adjusting the nitrogen pressure regulators immediately upstream.

Item (33) is a remote pressure sensor data desired by McDonnell Douglas for CRT display. Basic McDonnell Douglas data are included in Display 7, Figure 139, for use by the Lockheed system operator when also serving as the consolette operator.

It is also recommended that lighted switches for valves H through J be provided on the McDonnell Douglas console to indicate the position of the associated shutoff valves.

5.7.6 Purge System

5.7.6.1 Purge System Instrumentation

The aircraft purge system illustrated schematically in Figure 140 provides purge air to both the Lockheed and McDonnell Douglas suction systems to prevent/remove any cleaning liquid that might accumulate in or contaminate the suction systems. This common purge-air supply is provided from two separate sources that cannot be used simultaneously. The aircraft system should be exclusively controlled from the consolette with specified displays available to both the Lockheed and McDonnell Douglas console operators.

The primary purge system operates off high-pressure compressor bleed air from the JT12 primary propulsion engines and is for use at altitudes above 3,658m (12,000 ft) only. The second source of purge air is for altitudes below 3,658m (12,000 ft) and is supplied from the aircraft air-conditioning system. These systems have a common segment in which the system monitoring instrumentation is located. This instrumentation, Figure 140, items (40) through (43), is located in the cabin. The instrument at (40) is a dial-indicating pressure gauge located on the McDonnell Douglas console and sensing line pressure immediately downstream of the pressure regulator. A remote pressure sensor is located at (41). Item (42) is a temperature switch set for 68.3°C (155°F). Instrument item (43) is an overpressure warning switch set to actuate a green light at pressures below 100 $\pm$ 3.4 KPa (14.5 $\pm$ 0.5 psig), and a red light if the pressure exceeds this value. Instrument item (44) is a switch mounted on the cabin air modulating valve set to indicate that the valve is within 0.17 rad (10 deg) of the full-open position. These switches actuate lights are located on the McDonnell Douglas console. The requirements for this instrumentation are included in Table 13. This instrumentation is provided primarily for monitoring the system and to inform the Lockheed and McDonnell Douglas console operators of the availability of purge air. A spring-loaded-neutral switch is located on the McDonnell Douglas console to modulate the purge air temperature which may be monitored by observing the chamber valve temperature (3) on CRT Display 1. This switch is paralleled with a cockpit switch used to control emergency pressurization air temperature.

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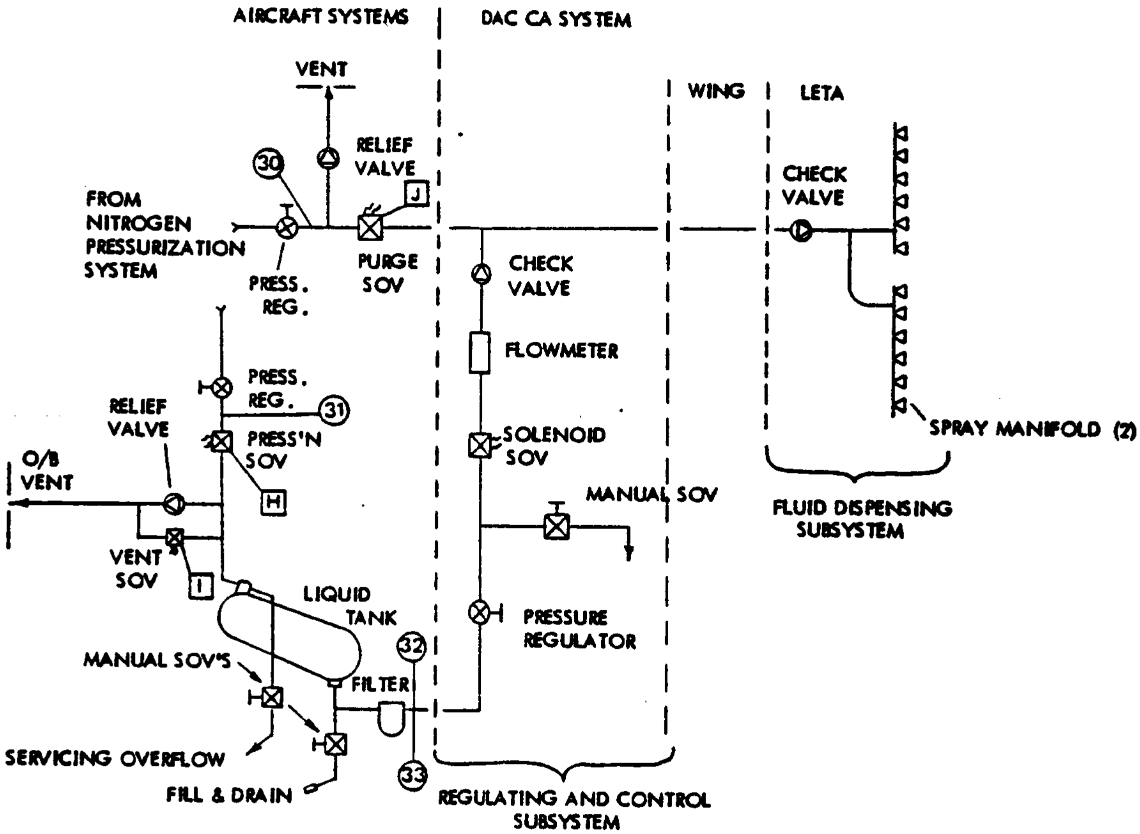

Figure 138. McDonnell Douglas Contamination Avoidance System

TABLE 12. McDONNELL DOUGLAS CONTAMINATION AVOIDANCE LIQUID SUPPLY INSTRUMENTATION

KEY NUMBER ON FIGURE 138	PARAMETER	NUMBER OF ITEMS	UNITS	RANGE	ACCURACY
30	NITROGEN PURGE SUPPLY PRESSURE TO LIQUID LINE (DIAL PRESSURE GAUGE IN AFT COMPARTMENT)	1	PSID	0 - 350	± 1%
31	NITROGEN SUPPLY PRESSURE TO LIQUID TANK (DIAL PRESSURE GAUGE IN AFT COMPARTMENT)	1	PSID	0 - 350	± 1%
32	LIQUID SUPPLY PRESSURE TO DAC SYSTEM (DIAL PRESSURE GAUGE IN AFT COMPARTMENT)	1	PSID	0 - 350	± 1%
33	LIQUID SUPPLY PRESSURE TO DAC SYSTEM (FOR DATA RECORDING AND CRT DISPLAY)	1	PSID	0 - 350	± 1%

McDonnell Douglas Contamination Avoidance Liquid System Displays

The displays recommended for McDonnell Douglas are only provided to supplement their desired data with required data for the liquid supply system and for ground setting of the supply system. Items (30), (31), and (32) in Figure 138 are dial-indicating pressure gages and should be so located that they may be observed while adjusting the nitrogen pressure regulators immediately upstream.

Item (33) is a remote pressure sensor data desired by McDonnell Douglas for CRT display. Basic McDonnell Douglas data are included in Display 7, Figure 139, for use by the Lockheed system operator when also serving as the consolette operator.

It is also recommended that lighted switches for valves H through J be provided on the McDonnell Douglas console to indicate the position of the associated shutoff valves.

5.7.6 Purge System

5.7.6.1 Purge System Instrumentation

The aircraft purge system illustrated schematically in Figure 140 provides purge air to both the Lockheed and McDonnell Douglas suction systems to prevent/remove any cleaning liquid that might accumulate in or contaminate the suction systems. This common purge-air supply is provided from two separate sources that cannot be used simultaneously. The aircraft system should be exclusively controlled from the consolette with specified displays available to both the Lockheed and McDonnell Douglas console operators.

The primary purge system operates off high-pressure compressor bleed air from the JT12 primary propulsion engines and is for use at altitudes above 3,658m (12,000 ft) only. The second source of purge air is for altitudes below 3,658m (12,000 ft) and is supplied from the aircraft air-conditioning system. These systems have a common segment in which the system monitoring instrumentation is located. This instrumentation, Figure 140, items (40) through (43), is located in the cabin. The instrument at (40) is a dial-indicating pressure gauge located on the McDonnell Douglas console and sensing line pressure immediately downstream of the pressure regulator. A remote pressure sensor is located at (41). Item (42) is a temperature switch set for 68.3°C (155°F). Instrument item (43) is an overpressure warning switch set to actuate a green light at pressures below 100 ± 3.4 KPa (14.5 ± 0.5 psig), and a red light if the pressure exceeds this value. Instrument item (44) is a switch mounted on the cabin air modulating valve set to indicate that the valve is within 0.17 rad (10 deg) of the full-open position. These switches actuate lights are located on the McDonnell Douglas console. The requirements for this instrumentation are included in Table 13. This instrumentation is provided primarily for monitoring the system and to inform the Lockheed and McDonnell Douglas console operators of the availability of purge air. A spring-loaded-neutral switch is located on the McDonnell Douglas console to modulate the purge air temperature which may be monitored by observing the chamber valve temperature (3) on CRT Display 1. This switch is paralleled with a cockpit switch used to control emergency pressurization air temperature.

CLEANING/ANTI-ICING	
TANK PR	XX.XX (24)
LIQUID MANIFOLD PR	XX.XX (21)
LIQUID LINE TEMP	XXX (23)
LIQUID FLOW RATE	XX.XX (22)
SLOT LIQUID TEMP	XXX (20)
DAC LIQUID SUPPLY PR	XXX.X (33)
DAC LIQUID FLOW RATE	XX.XX (DAC)

PURGE	SOURCE XXXXXXXXX		SET CODE XXXXX
PURGE LINE DP CAB	XX.XX (41)		
PURGE LINE DP AMB	XX.XX (41)		
CHAM PR VALVE	U	L	D
DES POS	XXX.X	XXX.X	
ACT POS	XXX.X (7)	XXX.X (7)	XXX.X (7)
DES PR DP AMB	XX.XX	XX.XX	
PR DP AMB	XX.XX (5)	XX.XX (5)	XX.XX (5)
PR DP CAB	XX.XX (5)	XX.XX (5)	XX.XX (5)
TEMP	XXX.X (3)	XXX.X (3)	XXX.X (3)

Figure 139. Lockheed Cleaning/Anti-Icing System-Display 7

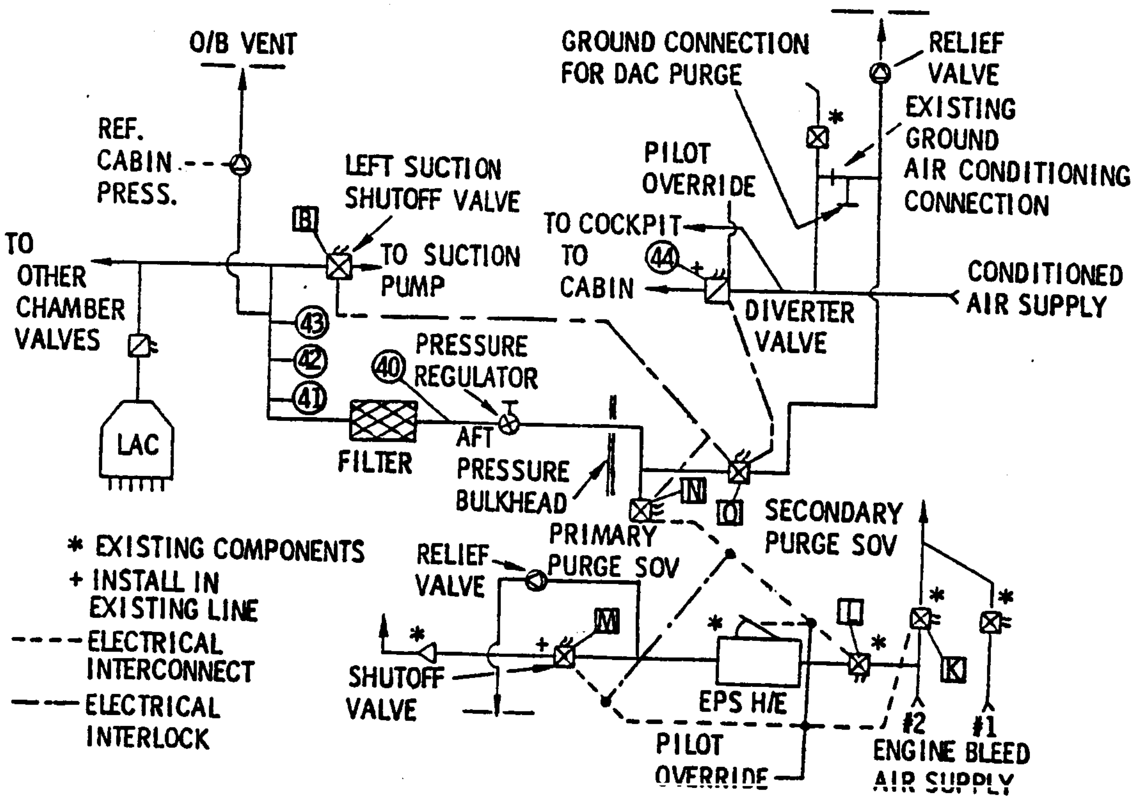

Figure 140. Purge System Schematic

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TABLE 13. PURGE SYSTEM INSTRUMENTATION

KEY NUMBER ON FIGURE 140	PARAMETER	NUMBER OF ITEMS	UNITS	RANGE	ACCURACY
40	PURGE FLOW REGULATED PRESSURE (DIAL GAUGE)**	1	PSID*	0 TO 25	± 1%
41	PURGE LINE PRESSURE	1	PSID*	0 TO 15	± 1%
42	OVERTEMPERATURE SWITCH (FOR WARNING LIGHT)**	1	LIGHT	155°F SET	± 5°
43	PURGE OVERPRESSURE SWITCH (FOR WARNING LIGHT)**	1	PSID*	14.5 SET	± .5 PSID
44	CABIN AIR MODULATING VALVE SENSOR (FOR LIGHT INDICATING VALVE "OPEN")**	1	LIGHT	10° SET	± 1°

*RELATIVE TO CABIN PRESSURE
**LOCATED ON DOUGLAS CONSOLE

Due to the relatively high pressure levels in the system during purge, it may be necessary to protect the Lockheed suction duct internal pressure transducers, 2 and 2a, from being subjected to this overpressure. NASA/DFRF will provide a system to isolate pressure sensors from overpressure conditions which could adversely affect the sensors.

5.7.6.2 Purge System Displays

The purge line pressure, (41), should be included on both the Lockheed and McDonnell Douglas data recordings on the PCM system. The data should also be available to both console operators for CRT display as a guide during purge system adjustments of the individual systems. It will be used for system monitoring by the operator of the McDonnell Douglas console from which the aircraft system is controlled. During the purge operation, each console operator will display data on the CRT from instrumentation in the suction system for which they are the controller.

Lockheed Purge Display

Display 7, Figure 139, for the cleaning/anti-icing system also includes the purge system display. This display includes purge line pressure differentials both relative to ambient and relative to the cabin for monitoring. Actual chamber valve data for temperature (3), pressure (5) and chamber control valve position (7) are also included. Also shown are desired chamber pressure control valve position and desired chamber pressure for the Lockheed system. These should be available from stored data for the input set code. Displays 2 and 3, Figures 132 and 133, should also be available for simultaneous displays while making purge system settings and adjustments.

LFC Systems Operator Displays

The LFC systems operator is of necessity stationed at the McDonnell Douglas console since this is where the controls and indicators are located except for the pump oil temperature indicator. The systems operator should have item (41) and chamber temperatures (3) available for setting the purge system. These are included in Lockheed Display 7 and should be included in the corresponding McDonnell Douglas display. In addition, the LFC systems operator is responsible for observing and setting the pressure regulator by observing the gauge indicated as (40) in Figure 140. The McDonnell Douglas console also includes lighted switches for valves (K) through (O) in Figure 140 and located on the McDonnell Douglas console. The color convention for these lighted switches has been selected to provide a green light for normal aircraft operation and an amber light for the purge operation alternatives. These switch lights should be green, as follows:

Valve	Valve Position(s)	Switch
K & M	Open	Purge Bleed Air (common)
L & N	Closed	Primary Purge (common)
O	Closed	Cabin Air Purge

Instrument items (42), (43) and (44) will indicate operational status by actuating lights on the McDonnell Douglas console. Item (42) will light a green light any time the line temperature is below 68.3 $\pm$ 3°C (155 $\pm$ 5°F), and red if the temperature exceeds this value. Item (43) will light a green light any time the purge air supply pressure is below 100KPa (14.5 psig), and will light a red light any time the purge air supply pressure is above 100KPa (14.5 psig). Item (44) will indicate that the cabin air modulating valve is full-open by lighting a green light and will indicate that the valve is partially closed by lighting an amber light.

5.7.7 Nitrogen System

The nitrogen system instrumentation is shown schematically in Figure 141. Instrumentation items (25), (30), and (31) have been discussed previously. Item (50) is a direct reading 0 - 2758KPa (0-400 psig) dial indicator located on the McDonnell Douglas console. This gauge is to be used for pre-flight setting of the nitrogen manual pressure regulator and can be monitored in flight by the LFC systems operator. Item (51) is a pressure switch set to actuate at 2,551 $\pm$ 69KPa (370 $\pm$ 10 psig) to light a green light on the McDonnell Doublas console at pressures below 2,551 $\pm$ 69KPa and to light a red light at pressures above 2,551KPa (370 psig).

Instrumentation item (52) is a pressure gauge to be used by the NASA/DFRF for instrument system purge, and the type of instrument and its range are at the discretion of the NASA/DFRF. The nitrogen shutoff valve (P) should have a lighted switch, mounted on the McDonnell Douglas console, with a green light for the closed position and an amber light for the open position.

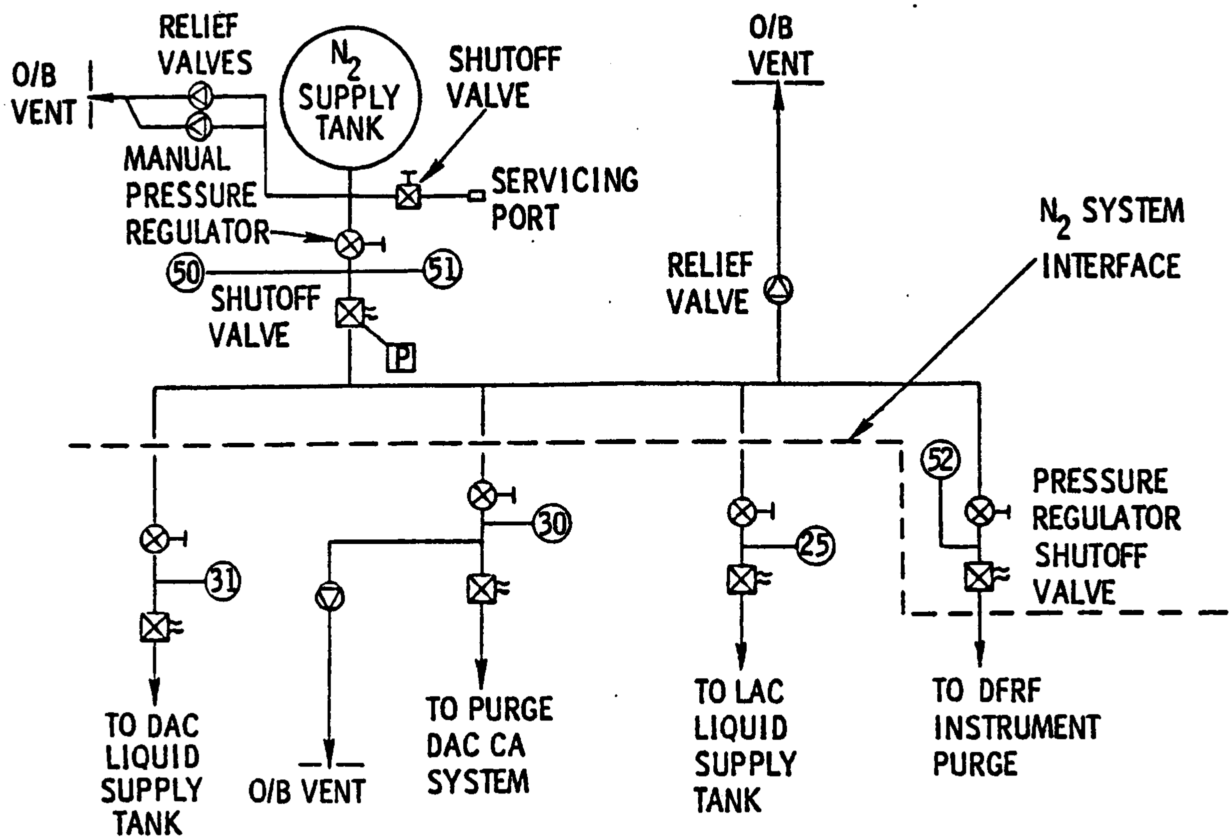

Figure 141. Nitrogen Pressurization System

5.7.8 Test Surface

5.7.8.1 Instrumentation

The surface instrumentation consists of static pressure data to obtain surface C_p distributions and to assist in the evaluation of slot spanwise flow distribution, surface hot film gages to determine chordwise laminarization characteristics, and a spanwise total pressure rake to determine the spanwise distribution of laminarization over the leading edge. Since the instrumentation is totally independent of the McDonnell Douglas test article, only the Lockheed surface instrumentation is addressed herein. All surface pressure instrumentation must have provision to be purged during or after liquid cleaning and suction system purge operations or while operating in rain.

5.7.8.2 Leading-Edge Test Article Instrumentation

All instrumentation on the surface of the leading-edge test article was installed during fabrication. The instrumentation pressure tubing and electrical wiring are routed to the inboard end of the test article, where the NASA/DFRF becomes responsible. The leading-edge surface instrumentation is illustrated schematically in Figure 142 and defined in Table 14. The instrumentation identified as (1) at A and E in Figure 142 (and also shown in

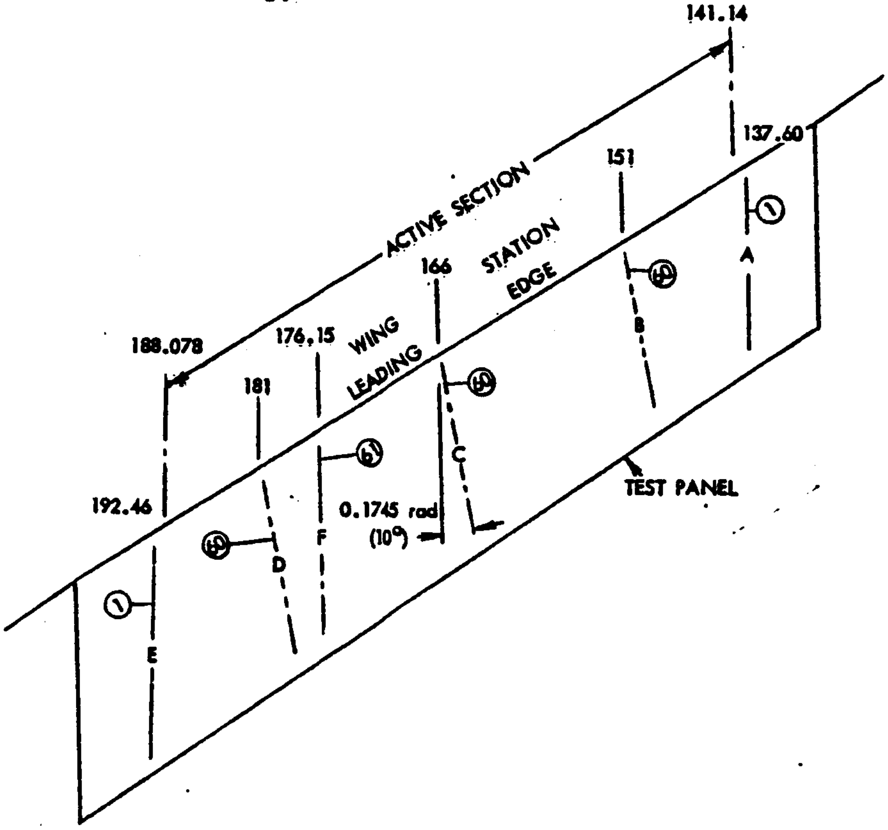

Figure 142. LETA Instrumentation

TABLE 14. LEADING EDGE SURFACE INSTRUMENTATION

KEY NUMBER ON FIGURE 142		NUMBER OF ITEMS	UNITS	RANGE	ACCURACY
1	SLOT AND SURFACE STATIC PRESSURE	44	PSID	-1.5 TO +1.5	± 1%
60	SUCTION SURFACE STATIC PRESSURE	42	PSID	-1.5 TO +1.5	± 1%
61	SUCTION SURFACE HOT FILM	6	*	*	*

*FURNISHED BY NASA

Figure 131) consists of rows of surface static taps inboard and outboard, respectively, of the active suction test surface. These taps are aligned spanwise with selected slots to provide a chordwise surface C_p distribution at these two locations. The fabrication and location of these pressure taps is described in Section 4.1.3.2. The taps are routed to the inboard leading-edge test article interface using instrumentation tubing.

There is a total of 44 of these surface pressure taps located adjacent to the active suction surface to aid in determining the C_p distribution on the LETA. Sixteen of these taps also serve as the reference pressure for the direct measurement of internal suction duct pressure (2) as shown in Figure 131. These 16 pressure instrumentation lines, (1), (8 on each end) must, therefore, be "teed" to connect with the ΔP (1)-(2) transducers and to a second transducer to measure the surface pressure. As noted in Sections 5.7.4.1 and 5.7.6.1, it is necessary to protect the ΔP (1)-(2) transducers from overpressure during purge operation. It should be noted that two of the 44 surface pressure taps and three of the internal collector duct pressure taps were found to be plugged upon arrival at NASA DFRF. These taps are not expected to be available for pressure measurements.

Instrumentation items (60) located at stations B, C, and D in Figure 142 are rows of surface static pressure taps. Each row consists of 14 taps with 7 each on the upper and lower surfaces. These are installed approximately midway between selected suction slot pairs on the active suction surface to evaluate the surface Cp distribution. The location and fabrication of these pressure taps is discussed in Section 4.1.3.2. Three of these surface taps were destroyed during repair of an adhesive-filled slot duct. The surface taps located aft of slot 6 at stations B, C, and D were lost during the stripping operation for repair of this slot. The ranges and other details of the surface pressure instrumentation are included in Table 14.

Instrumentation item (61) located at station F in Figure 142 is a row of 6 hot-film gages. These were provided and installed by NASA LaRC personnel. The hot-film sensors are spaced chordwise with two sensors located between slots U9 and U10, one sensor aft of slot U11, and one sensor between each of the following pairs of slots: U5 and U6, U8 and U9, and U10 and U11. The sensors are TSI Model 1268 flush-mount sensor elements having a diameter of 0.076 cm (0.030 in) and a length of 1.91 cm (0.75 in). Each sensor is encased in a stainless steel tube having a diameter of 0.15 cm (0.059 in). They are installed flush with the wing surface and within surface smoothness criteria to minimize possible delaminarizing effects. The two electrical connections to each hot-film sensor are routed to the inboard leading-edge test article interface by a coaxial cable with the transition to coaxial cable as close to the sensor as practicable. In addition to the gages, NASA LaRC will provide any special considerations, ranges, accuracies, procedures, and interfacing signal conditional requirements.

5.7.8.3 Aft Fairing Instrumentation

The recommended aft fairing instrumentation is shown schematically in Figure 143. Instrumentation item (63) is a row of 10 surface static pressure taps and item (64) is a row of 6 hot-film gages. The ranges of these measurements is the same as their counterparts on the leading edge in Table 14. This

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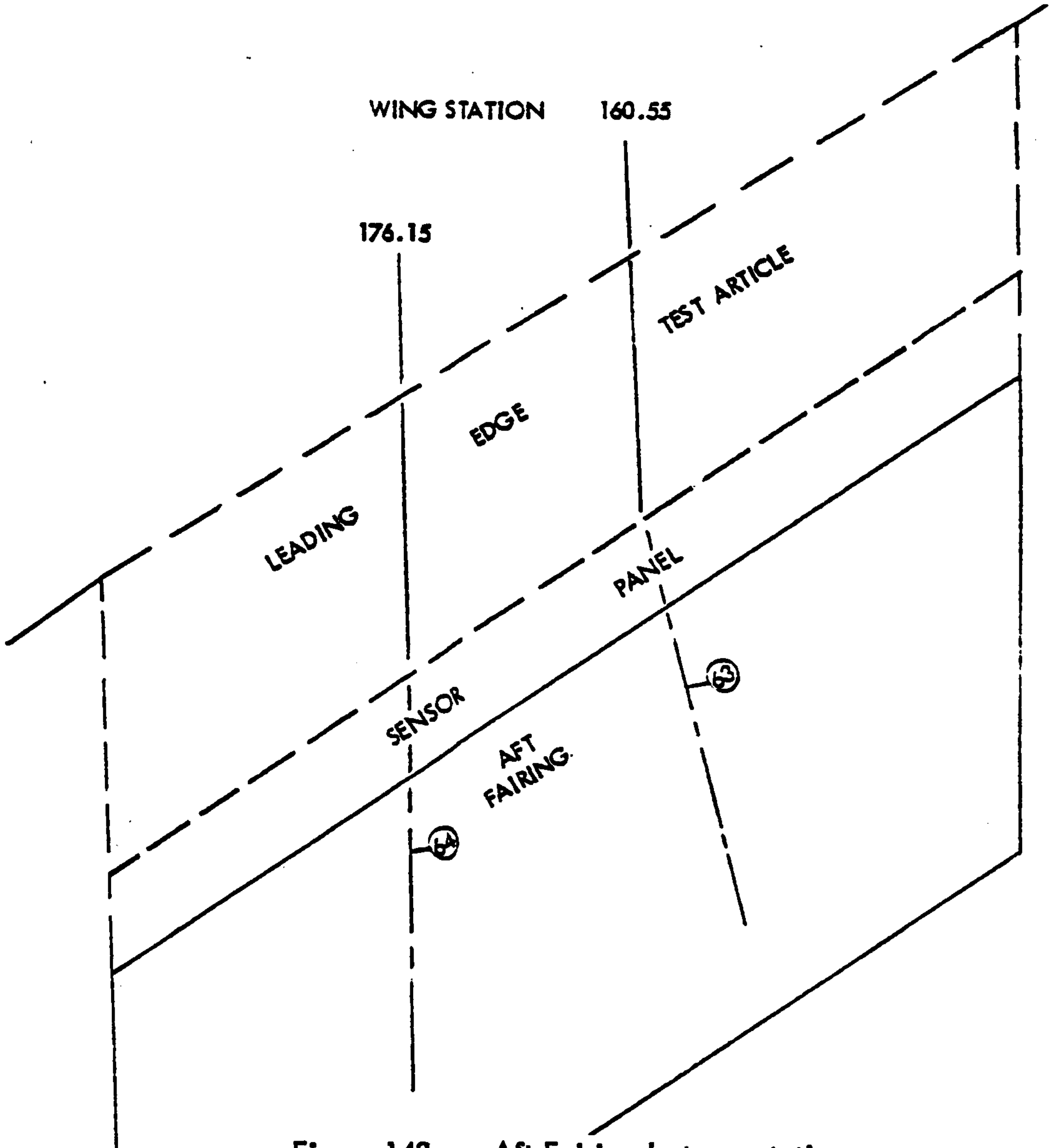


Figure 143. Aft Fairing Instrumentation

instrumentation is located and its identification called out on the appropriate Lockheed drawing. The hot-film sensors are located in-line chordwise with the hot-flim sensors on the test article, i.e. at wing station 176.15. They are spaced chordwise on 13.72 cm (5.4 in) centers with the first sensor located 9.53 cm (3.75 in) aft of 12 percent chord. The surface static pressure lines will require purging during all cleaning system and suction/cleaning systems purge operations or while operating in rain. The hot-film gages will be the same as those noted in the previous section and will be supplied by NASA LaRC. The installation of all instrumentation in the aft fairing is the responsibility of the NASA DFRF.

5.7.8.4 Preston Probe Spanwise Rake

The NASA will provide upper and lower surface total pressure rakes to mount along the leading edge attach lines by replacing selected leading edge attachment screws with chamfered washers and then securing the rakes with longer screws in the same locations. The tips of the probes should be at least 1.27 cm (0.5 in) chordwise aft of the last slot and uniformly spaced across the span. The total pressure probes should be at approximately 5.08 cm (2 in) intervals along the wing stations and aligned to project chordwise forward of the rake mounting bars. Each rake should have two free-stream total-pressure probes projecting outside any boundary layer influence. These free-stream probes should be used as the reference pressure for individual probe ΔP measurements. When installed, some of these pressures will be substituted in the data system for some of the surface static taps (1) at stations A and E in Figure 142, but not including those associated with the ΔP (1)-(2) measurements. These rakes and the particulars on sensor requirements are the responsibility of the NASA. These rakes will require purging of the pressure probes whenever the cleaning system or suction/cleaning sytem purge is in operation or when flying through rain or ice crystals.

5.7.8.5 Displays

The displays of the surface pressure instrumentation will be used as integrated displays and data from this instrumentation will also be used in computing some of the displays identified in Sections 5.7.4.4 and 5.7.4.5. Therefore, all surface pressure instrumentation will be discused without breakout for the component in which it is located. Due to the high Cp gradients in the extreme leading-edge region, chordwise data are displayed versus slot number as a means of expanding the chordwise scale in this region as shown in Display 8, Figure 144. This also maintains compatibility with the suction data displays of Section 5.7.4. An alternate optional display may be used versus x/c at the NASA's discretion. The following display capabilities are recommended:

(a) Chordwise Cp Distribution - Wing Station C should also have the option for a display versus x/c that is expanded to include the surface static pressure measurements (63) located on the upper-surface aft fairing extending to an x/c of approximately 42 percent. This display is similar to Display 8, except that Cp is a function of x/c.

(b) Spanwise Cp Distribution - Displays of the Cp distribution versus spanwise location are recommended for each instrumentation chordwise location. These should include the actual Cp data at wing stations B, C, and D with chordwise interpolations of the distributions of A and E as illustrated in Display 9, Figure 145.

(c) Chordwise Laminarization (Hot Film) - The hot-film data should be displayed with optional capability for displaying only the 6 hot-film gages on the leading-edge test article or all 12 hot-film gages including those on the aft surface. When the spanwise rake of Section 5.7.8.4 is installed, the hot-film data on the aft surface will be meaningless. The display should carry a reference to judge whether the boundary layer is laminar or turbulent. The display for the leading edge only should be versus slot number as illustrated in Display 10, Figure 146, while the display including those on the aft fairing should be versus x/c and correspond with (a) above.

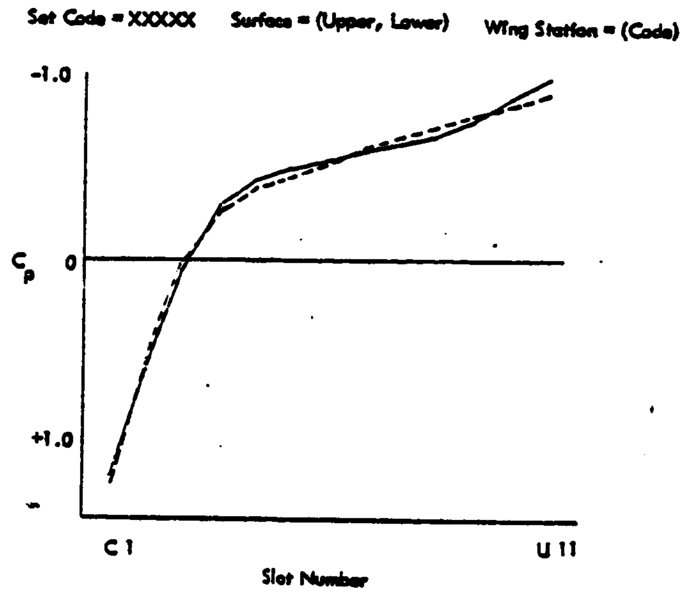


Figure 144. Chordwise Cp Distribution-Display 8

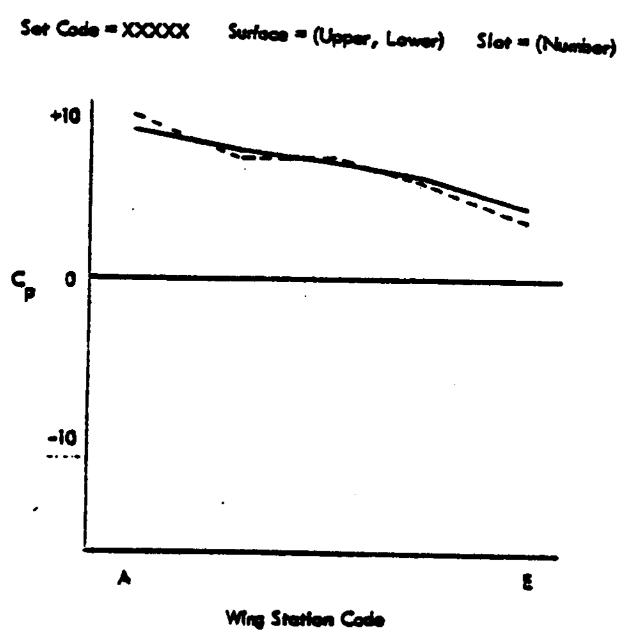
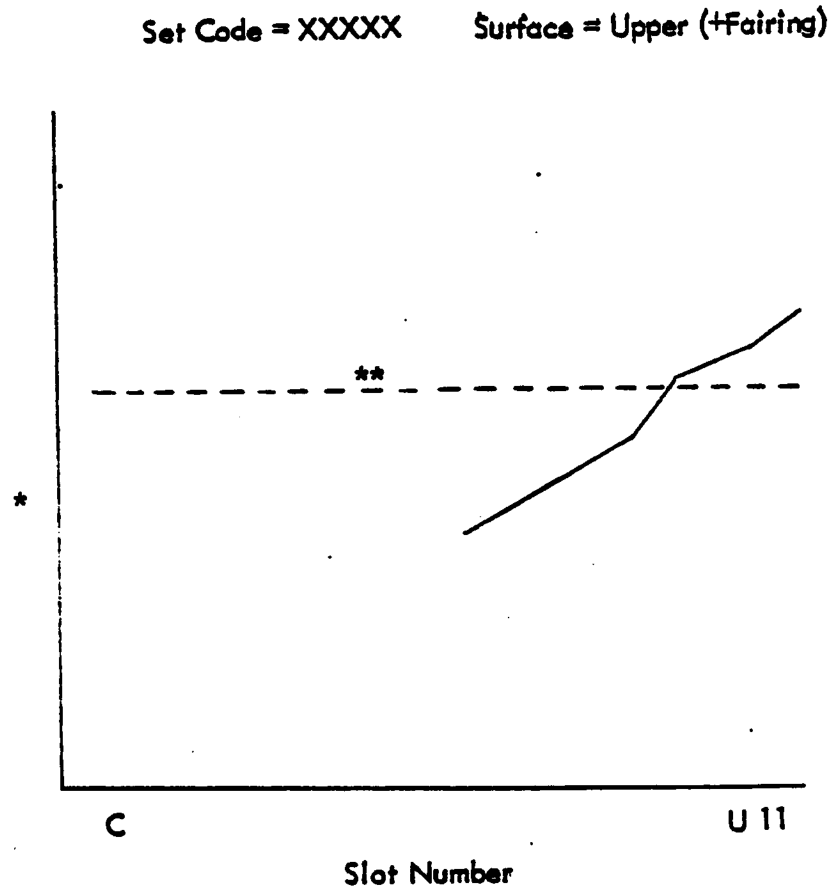


Figure 145. Spanwise Cp Distribution-Display 9



*Scale to be provided by NASA LaRC

**Criteria to be provided by NASA LaRC

Figure 146. Chordwise Laminarization-Display 10

(d) Spanwise Laminarization (Preston Rake) - The spanwise pressure ΔP data from the Preston rake should be displayed versus wing station with a reference line for judging whether the flow is laminar or turbulent as illustrated in Display 11, Figure 147.

5.7.9 Consoles

Separate consoles are to be designed and fabricated by the NASA/DFRF for the control of the Lockheed and McDonnell Douglas LFC systems. Originally, Lockheed recommended a third console or consolette to be designed and fabricated by the NASA/DFRF to provide control of the aircraft systems that are common to both the Lockheed and McDonnell Douglas LFC systems. This consolette was to include the AiResearch pump control panel and was to be demountable so that it may be mounted on either the Lockheed or McDonnell Douglas console and be operated by either console operator. It was determined later that the limited JetStar aisle space made a demountable control panel undesirable. As a result,

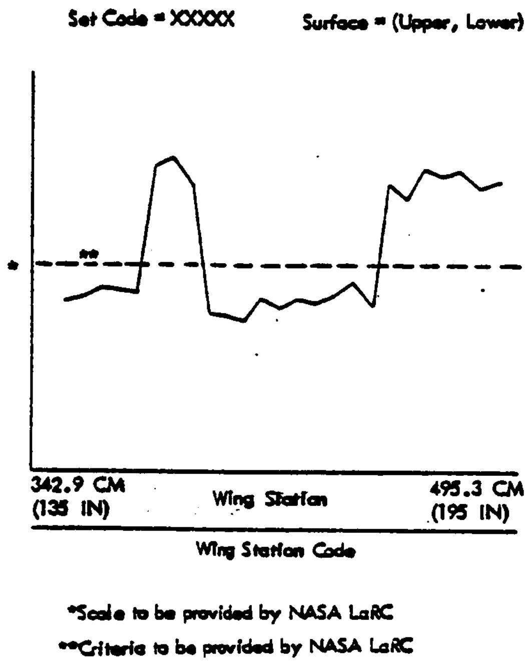


Figure 147. Spanwise Laminarization-Display 11

NASA DFRF personnel have designed the LFC systems and controls and the AiResearch pump control panel into the McDonnell Douglas console since it had more available space then the Lockheed console. Recommendations for the Lockheed console and for the LFC systems control portion of the McDonnell Douglas console are included herein. Recommendations for the remainder of the McDonnell Douglas console are the responsibility of McDonnell Douglas. The recommendations contained herein are flexible and may be altered to improve commonality; however, it is recommended that the functional characteristics be retained.

As far as possible, all valves are controlled by lighted switches with green indicating the normal aircraft conditions, and amber indicating that an LFC system is in operation. These switches are grouped so that a given system operation will be indicated by a group of amber lighted switches, while all other switches are lighted green. Some systems require warning of potentially dangerous operation and will be indicated by red lights; safe operation will be indicated by green lights.

5.7.9.1 McDonnell Douglas Console - LFC Systems Control/Indicators

A portion of the McDonnell Douglas console will be dedicated to aircraft LFC systems. These controls and indicators will provide the primary control for the aircraft systems common to both the Lockheed and McDonnell Douglas systems. They will include the pump control panel provided by AiResearch. The remainder of the controls are those recommended by Lockheed for control of the suction, nitrogen and purge systems. The complete array of the recommended controls and indicators is illustrated in Figure 148. The arrangement shown is for illustration only; the actual arrangement will be determined by NASA DFRF.

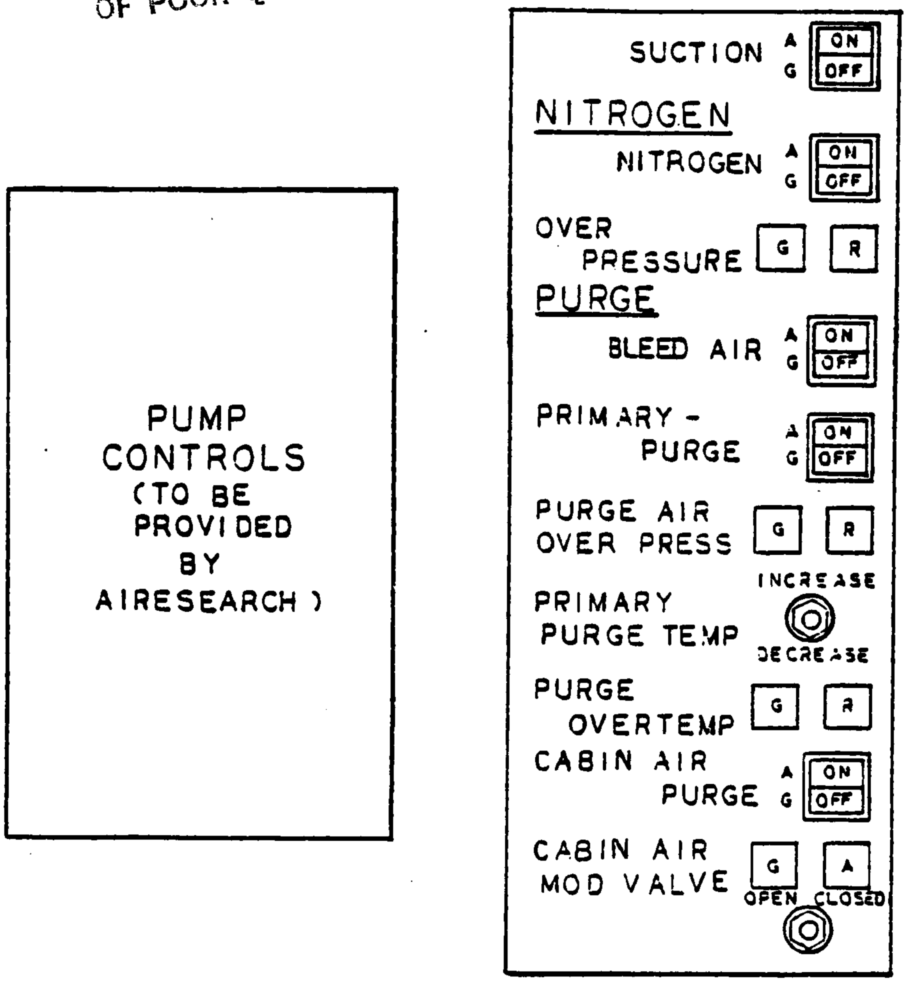


Figure 148. LFC Systems Controls - McDonnell Douglas Console

5.7.9.2 Lockheed Systems Control Console

The primary features of the Lockheed console are the CRT scope and keyboard and the 26 needle valve control potentiometers as shown in Figure 149. Both of these items have been or will be selected by the NASA. For purposes of this document, a VT103 CRT has been used in Figure 149 and 3.18 cm (1.25 in) diameter indicating knob potentiometers have been assumed for the needle valve controls. This arrangement is recommended for the convenience of the operator to keep the needle valve controls convenient for adjustment while observing the results of the adjustment on the CRT. The two chamber pressure control valves are controlled by spring-loaded neutral switches to drive the valves in the open or closed direction while observing the valve position and chamber pressure changes on the CRT. These valves may be used in either the suction or purge mode but will usually be driven to the full-open position in the suction mode.

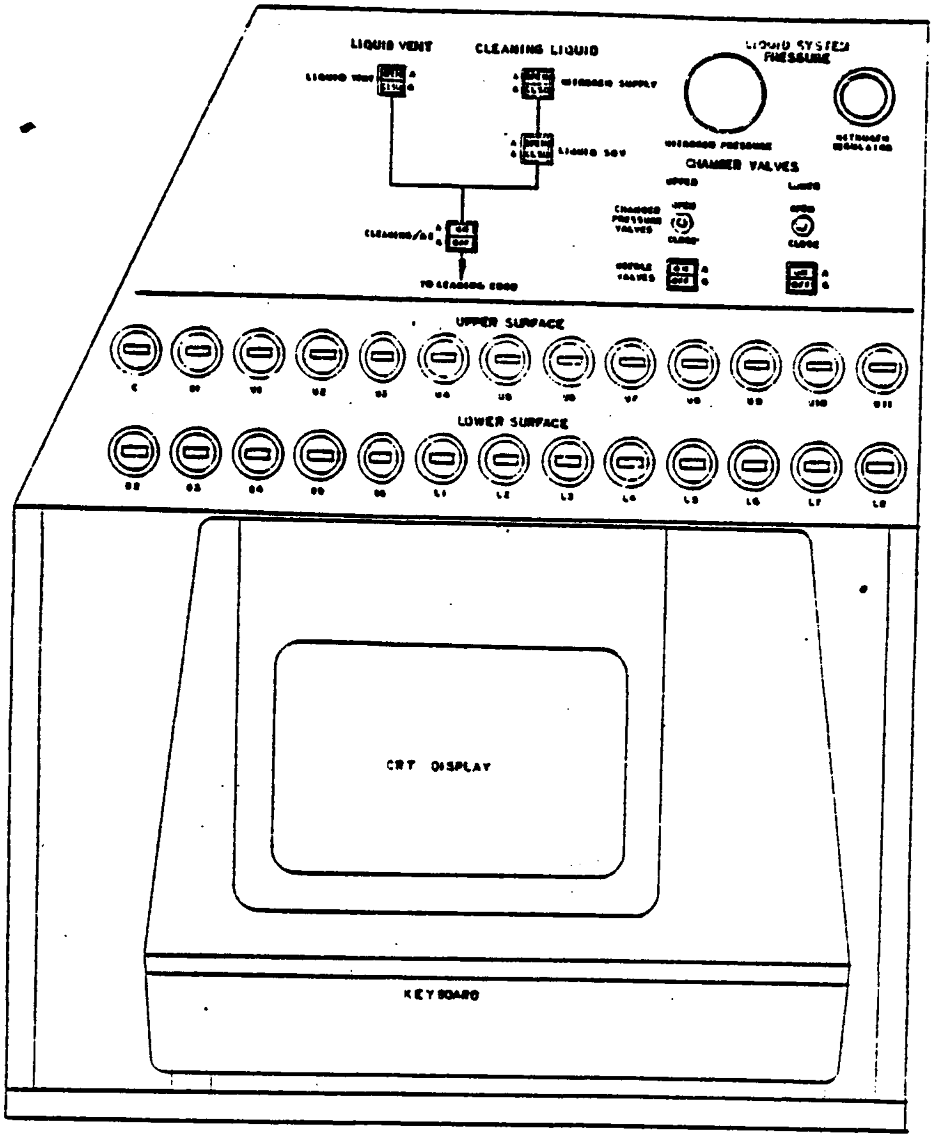


Figure 149. Lockheed Console

The nitrogen pressure regulator and adjacent pressure gage are provided to pre-set or adjust the cleaning liquid tank pressure and consequently the cleaning liquid flow. The remainder of the controls on the console control the cleaning system liquid supply and the cleaning system vent with a single switch operating the seven cleaning/suction 3-way valves for the dedicated cleaning and the dual-purpose slots, C1 and C2, and D1 through D6, respectively.

6.0 AIRCRAFT MODIFICATIONS

6.1 AIRFRAME

The NASA JetStar requires significant modification to incorporate the LFC-LEFT test articles and systems. Visible external changes to the aircraft are shown in Figure 150.

To accomplish the necessary wing rework for installation of the LFC test sections, the JetStar upper wing panels must be removed, which requires jacking and cradling of the aircraft to provide adequate support. Jacks, wing supports, and fuselage cradles are shown in Figure 151.

6.1.1 Wing Trailing Edge

When the JetStar external fuel tank is removed to provide room for installation of the test article, the landing lights that are in the nose of the fuel tank are lost. They are replaced by fold-down landing lights, one on each wing in the lower surface of the trailing edge. As shown in Figure 152, this is in the new wing trailing-edge structure which is added to fill the gap left by removal of the fuel tanks.

The added structure is similar to the structure already on the JetStar wing; i.e., aluminum ribs, stiffeners and skin. Figure 153 shows the lower trailing-edge rework with the landing light installation. Upon removal of the external fuel tank, a gap is left in the wing trailing-edge flaps. This gap is closed by adding a section to both the inboard and outboard flap panels. The

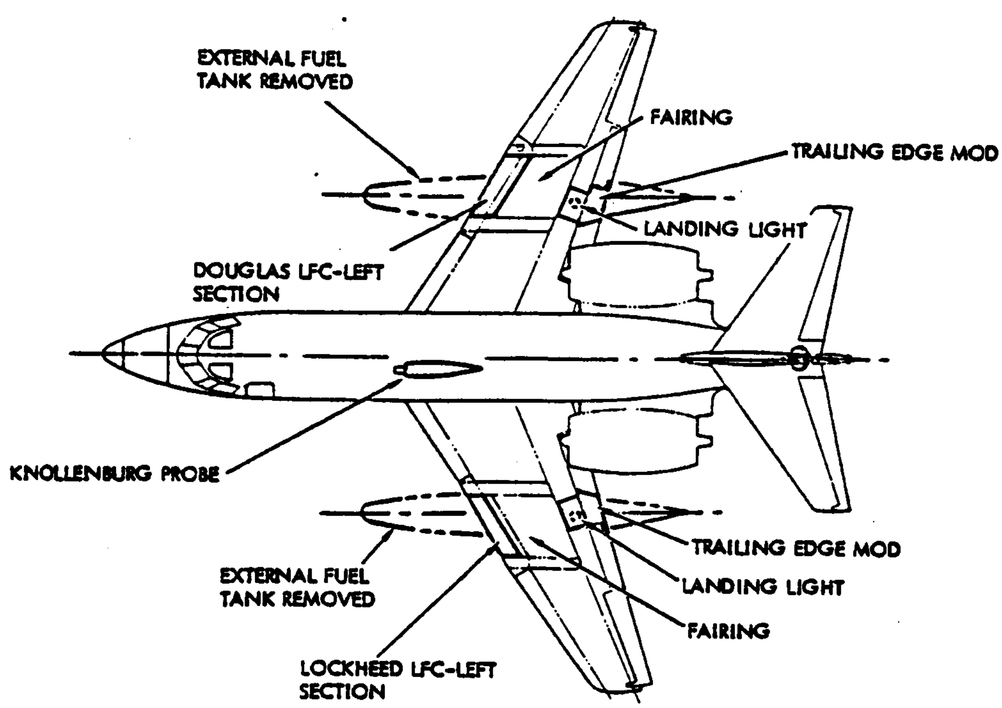


Figure 150. LFC-LEFT JetStar

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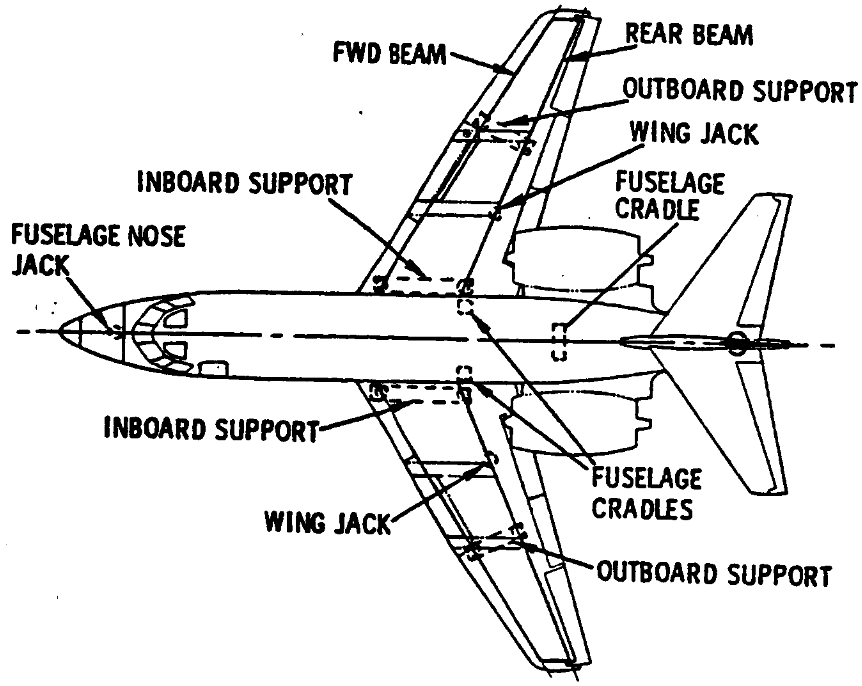

Figure 151. JetStar Jacking and Support

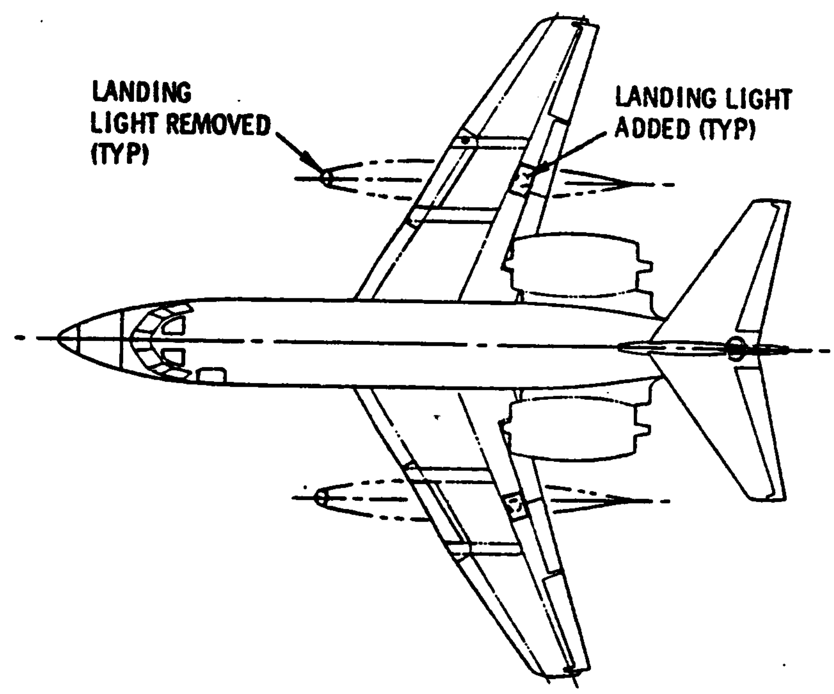

Figure 152. Trailing Edge Addition

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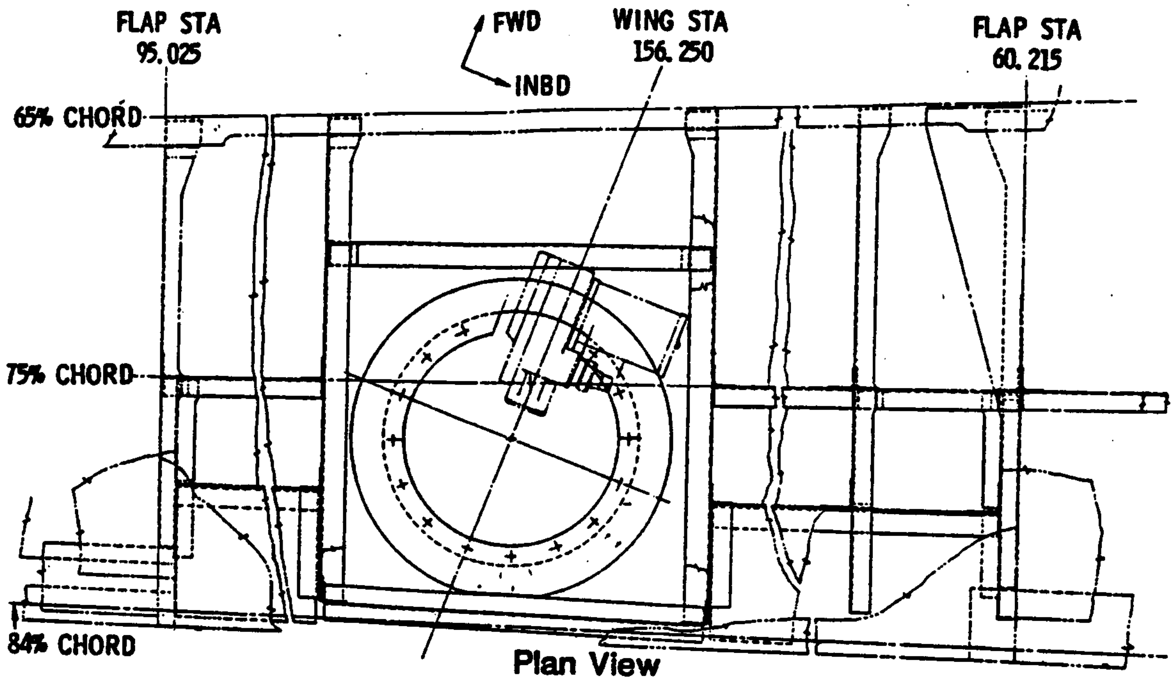

Figure 153. Wing Trailing Edge Structure

addition on each flap is sized to properly distribute the added air loads on the existing JetStar flap hinges. LFC required flap modification design takes into account a previous flap modification performed on the NASA JetStar. The basic trailing-edge flap of the JetStar has a slot in its leading edge that is not duplicated in the leading edge of the extension. Figure 154 shows the flap extensions.

6.1.2 Leading-Edge Flap

Existing JetStar leading-edge flaps are modified to accommodate the LFC test sections. The inboard-most flap section on each side is removed to accommodate the LFC test articles while the two outboard flap sections on each side are locked in the retracted position. Leading-edge flap extension is not used on the test airplane. With the leading-edge flaps inactive, the flap drive torque tubes are removed, and the flap drive motor is moved to the wing leading edge just outboard of the right-hand test article to drive the McDonnell Douglas shield installed in that test article. The leading-edge flap modification is shown in Figure 155.

The JetStar leading-edge flap drive motor and gear box is moved from the left hand wing leading edge root to the right hand test section outboard leading edge transition fairing. JetStar screw jacks and modified torque shafts are used to actuate the McDonnell Douglas shield. Figure 156 shows the drive motor gearbox installation.

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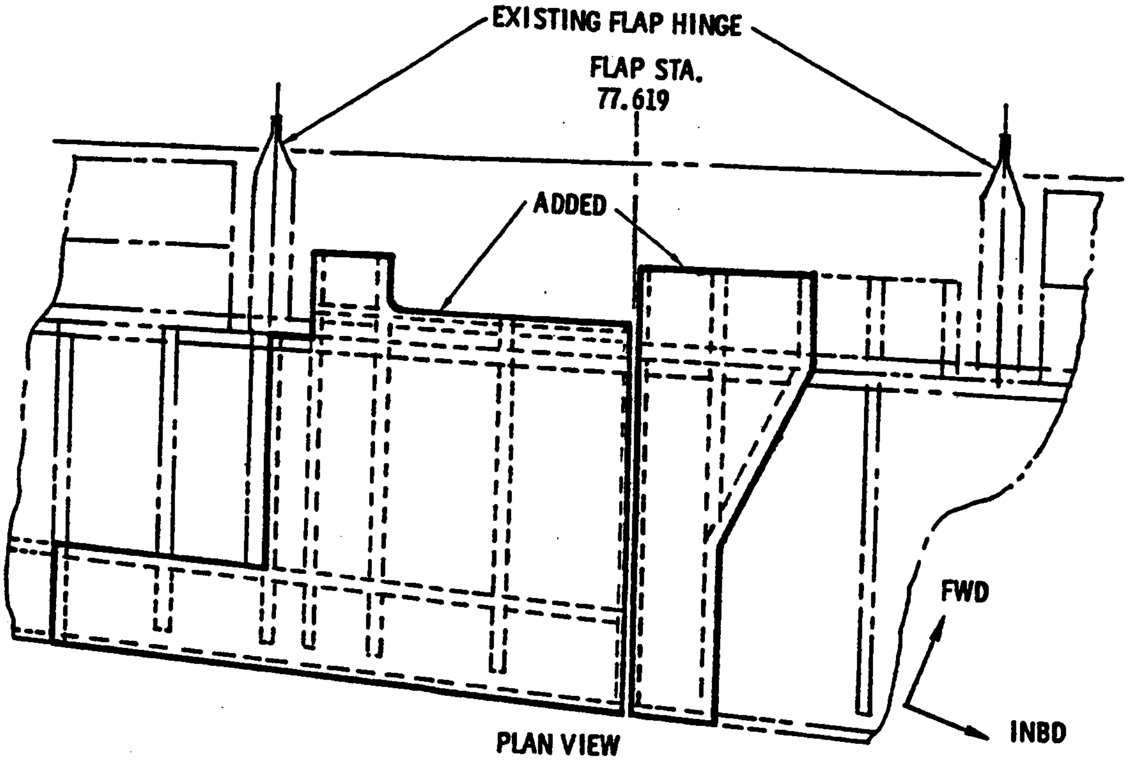

Figure 154. Trailing Edge Flap Additions

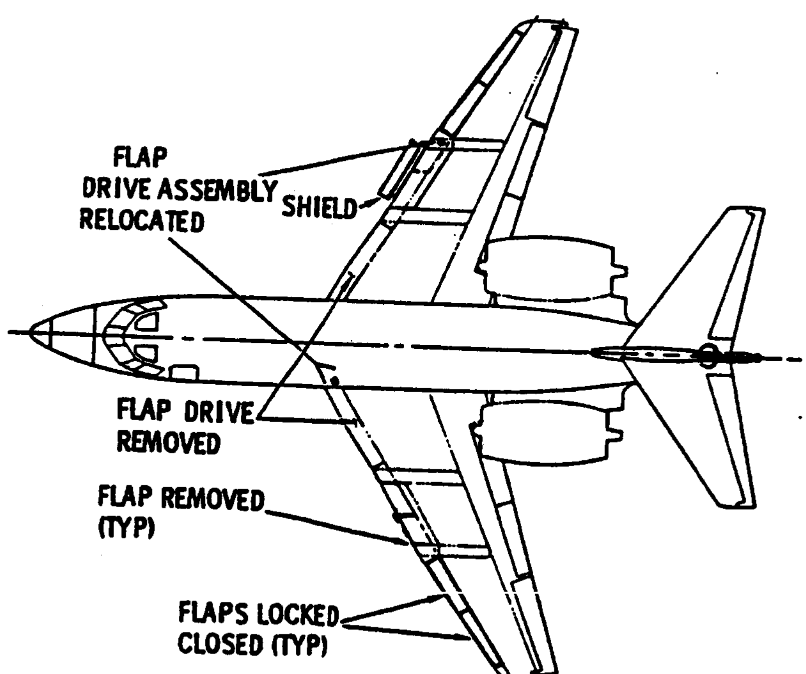

Figure 155. Leading Edge Flap Modification

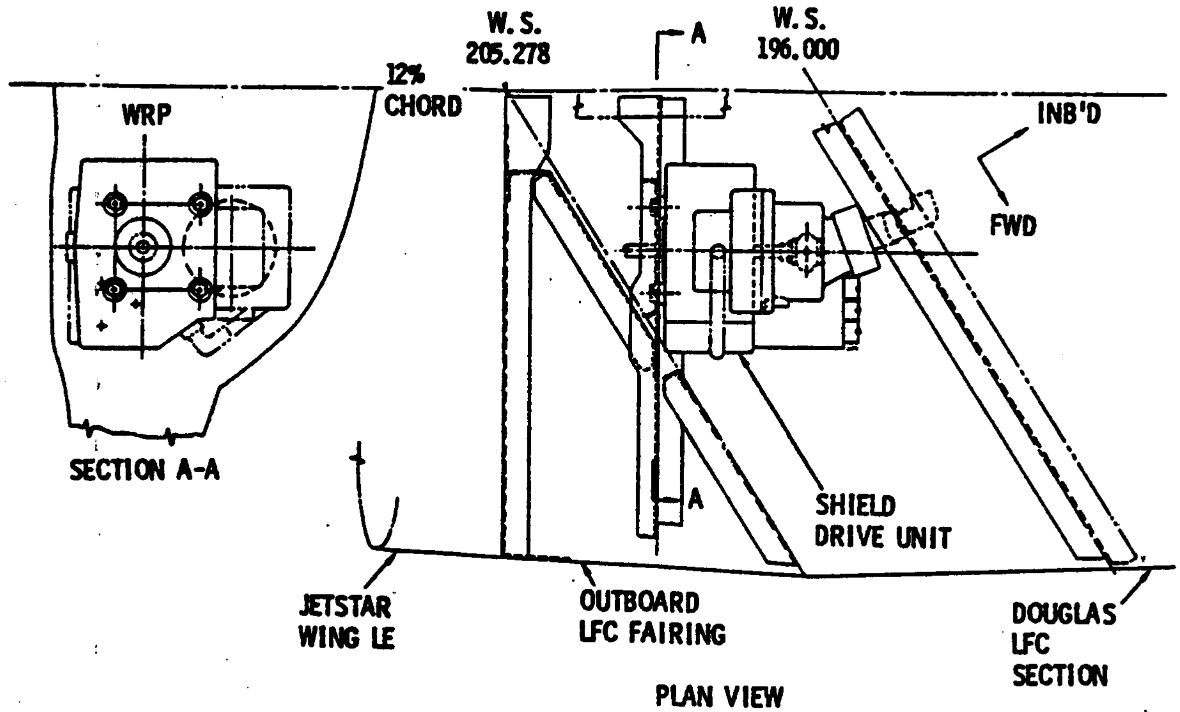

Figure 156. Shield Drive Motor Installation

6.1.3 Fixed Leading Edge

The JetStar wing leading-edge section between the test articles on each side and the fuselage are reworked to allow for the suction tube runs. These sections are a part of the normal wing inboard fuel tankage; however, for the LFC configuration these leading-edge sections become dry-bays.

Twenty-six suction lines run through the left-hand, fixed, leading-edge section, and 15 run through the right-hand section. These lines are shown in the approximate position they are located in the inboard end of the test articles in Figure 157.

To make a dry bay in the fixed leading edge it is necessary to seal the wing front beam by installing aluminum seal plates over the beam web lightening holes and some existing drain holes. Other drain holes are plugged with soft aluminum rivets. Sealant is applied to assure a fuel-tight seal.

Since the inboard section of the JetStar wing leading edge is not readily removable, the LFC modification work is accomplished with the leading edge in place. To facilitate this work, four additional access panels are cut in the leading-edge upper surface similar to an existing access panel. One fuel quantity probe is removed and an aluminum cover is installed over the opening. Figure 158 shows the front beam sealing and access panels in the leading edge surface.

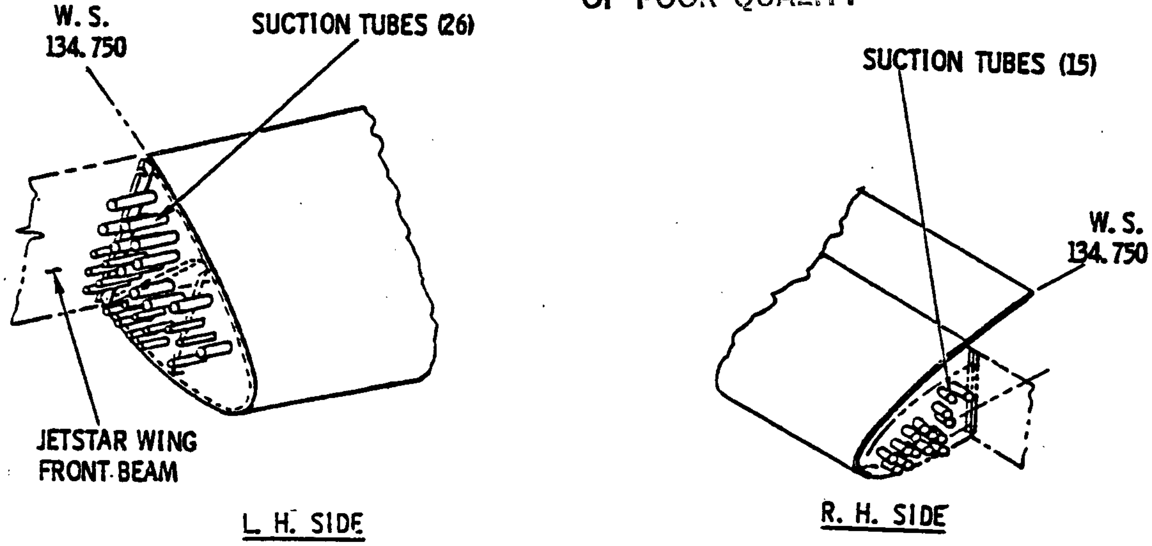

Figure 157. Test Article Suction Lines

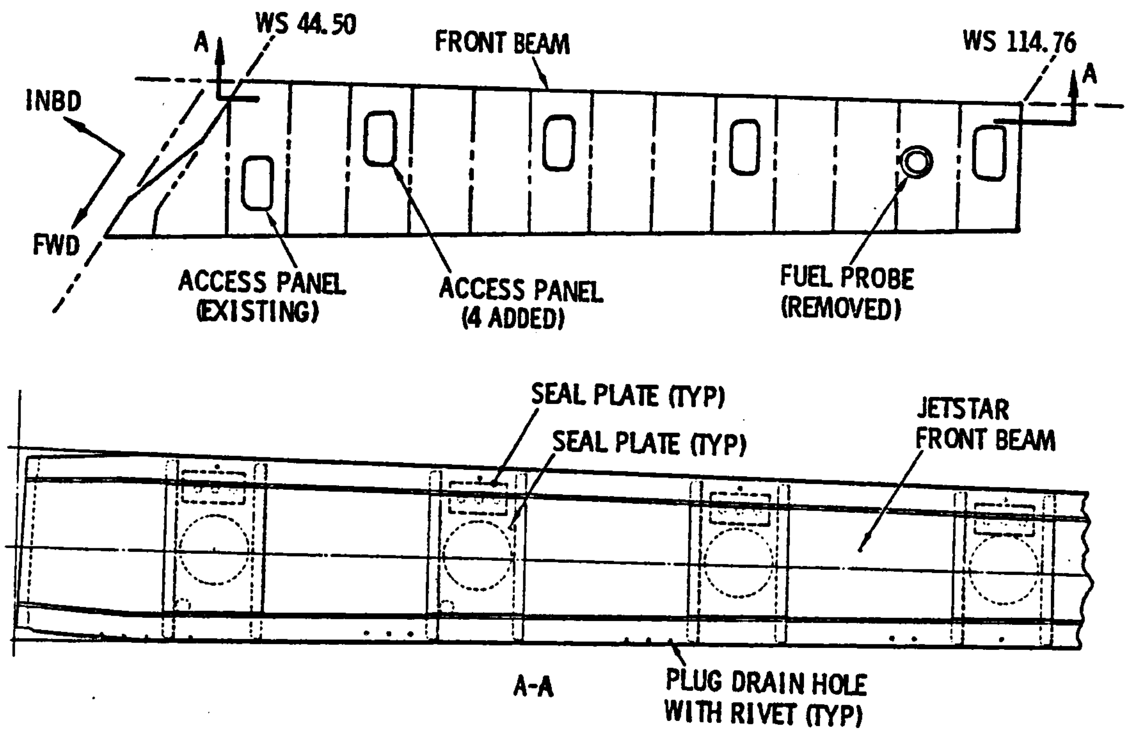

Figure 158. Wing Leading Edge Rework

End bulkheads and intermediate ribs in the leading-edge are reworked to accommodate installation of the suction tubes from each of the test articles. Holes are cut or enlarged in the outboard bulkhead and intermediate ribs as required with aluminum doublers and stiffeners added as necessary. On the inboard bulkhead, the forged aluminum plate normally installed on the JetStar for tank access is removed and a machined aluminum plate installed to accommodate the suction lines. Figure 159 shows some of the bulkhead and ribs rework. Existing intermediate formers between the bulkheads do not require rework for suction tube installation. Figure 160 shows typical suction, cleaning, and hydraulic tube installations in the fixed leading edge.

6.1.4 Fuselage LFC Line Entry

Provision for entry into the JetStar fuselage pressure shell for LFC suction lines, instrumentation lines, and electrical connections is made by the addition of adequate skin doublers and intercostals. The skin doublers and intercostals are made from aluminum sheet, and the necessary end clips are made from aluminum extrusions. Figure 161 shows a view of the left LFC-LEFT fuselage entry.

6.1.5 Fuel System

As a result of installation of the leading-edge test articles on the JetStar wings, some changes to the fuel system are required. The drying of the fixed leading edge revises the capacity of the inboard wing tanks which in turn requires recalibration of the fuel quantity system for these tanks. One fuel quantity probe is removed from the leading edge for the LFC installation, and

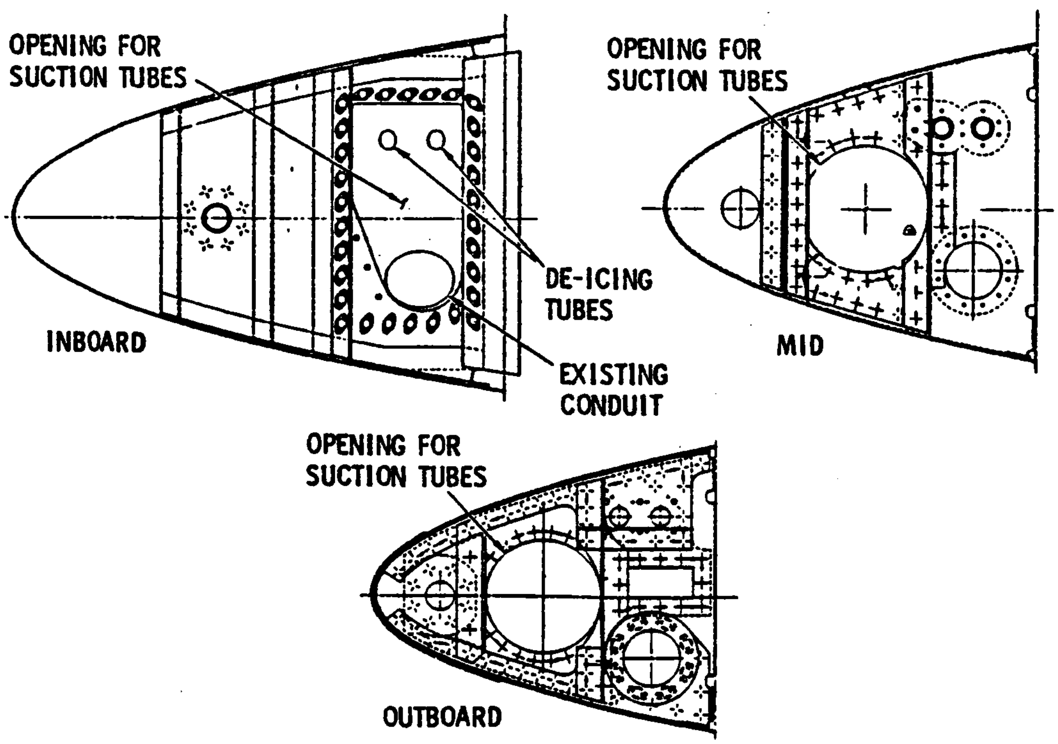

Figure 159. Rib Suction Tube Openings

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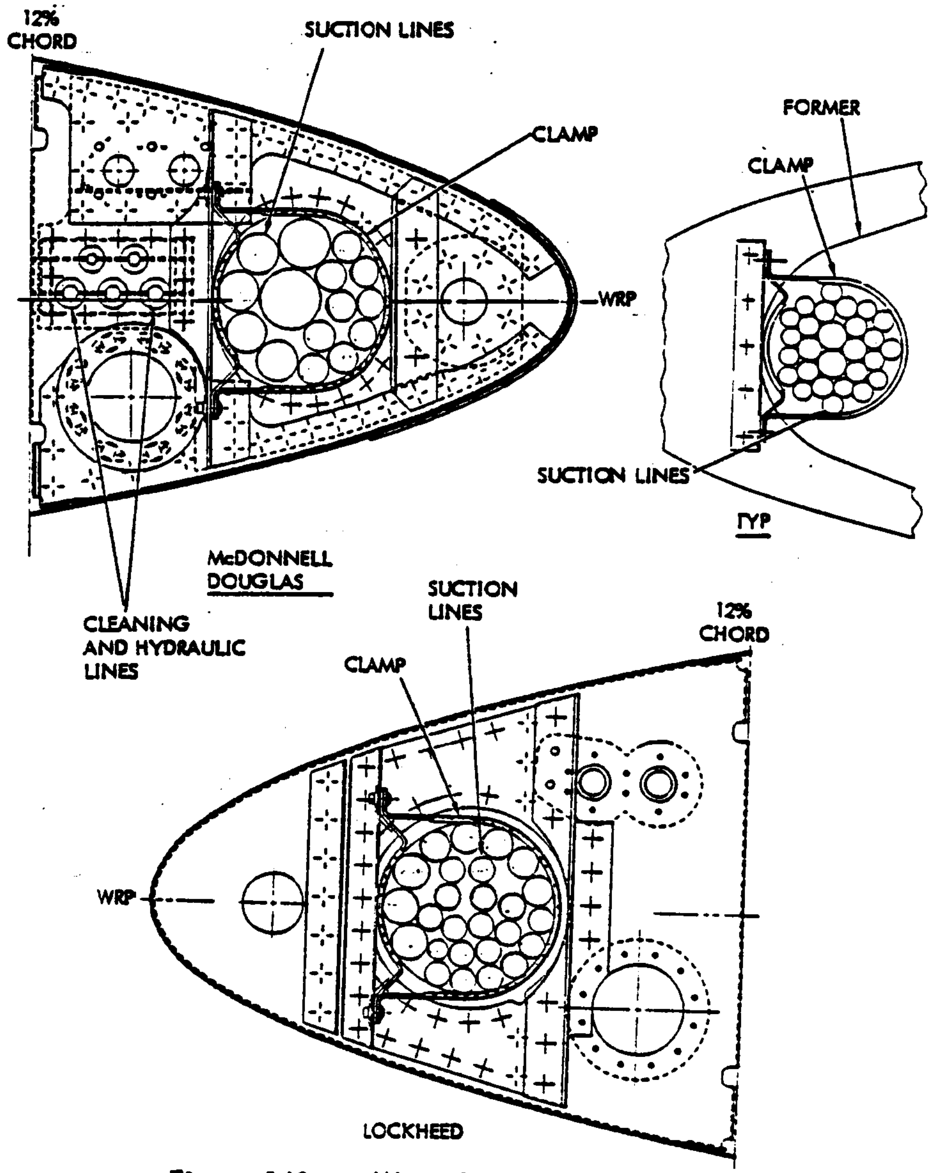

Figure 160. Wing Suction Tube Installation

one additional probe is removed from the main portion of the tank to help balance the system.

The single-point refueling system installed on the NASA JetStar is inactivated so that the servicing panel in the right-hand wing root is removed from the path of the LFC fuel suction lines. With the service panel removed, fuel servicing is accomplished by using the fillers on top of the wings. A servicing door is provided in the LFC fairing. Figure 162 shows the revised fuel system configuration.

Fuel quantity indicators for the inboard wing tanks are recalibrated to accommodate the changes required for the LFC configuration. Changes to the fuel quantity gauging systems are shown in Figure 163.

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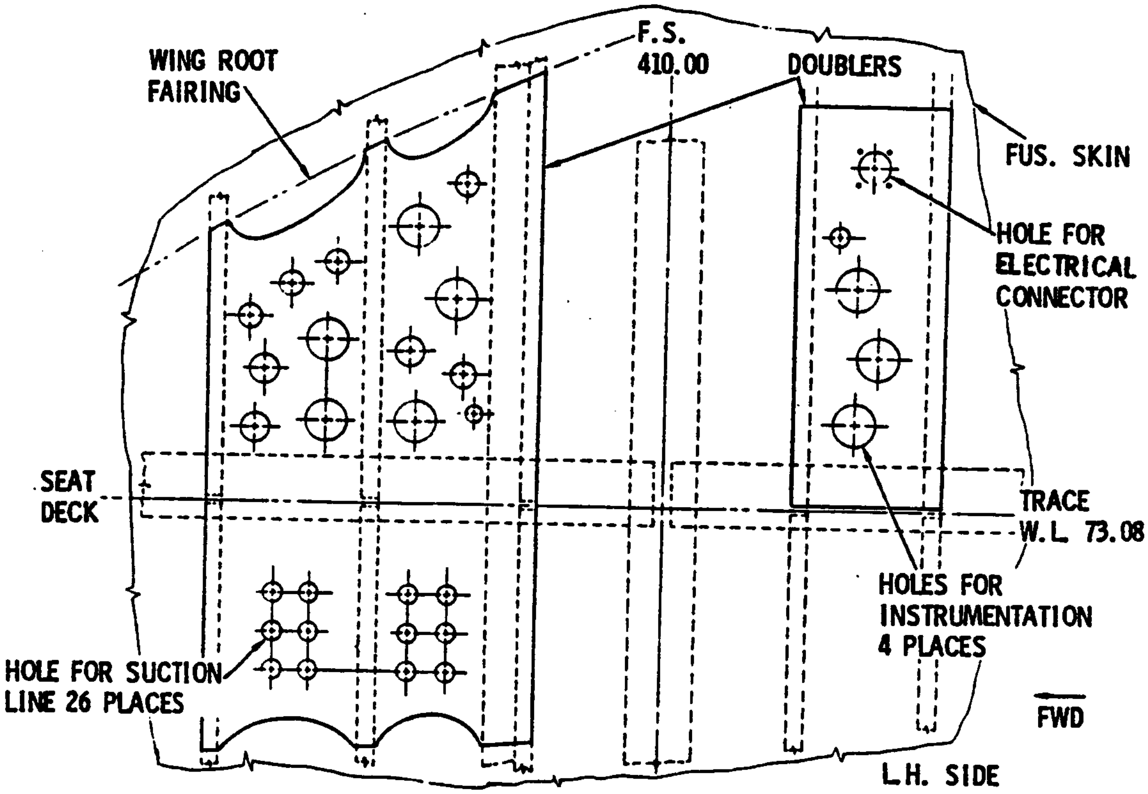

Figure 161. LFC-LEFT Fuselage Entry

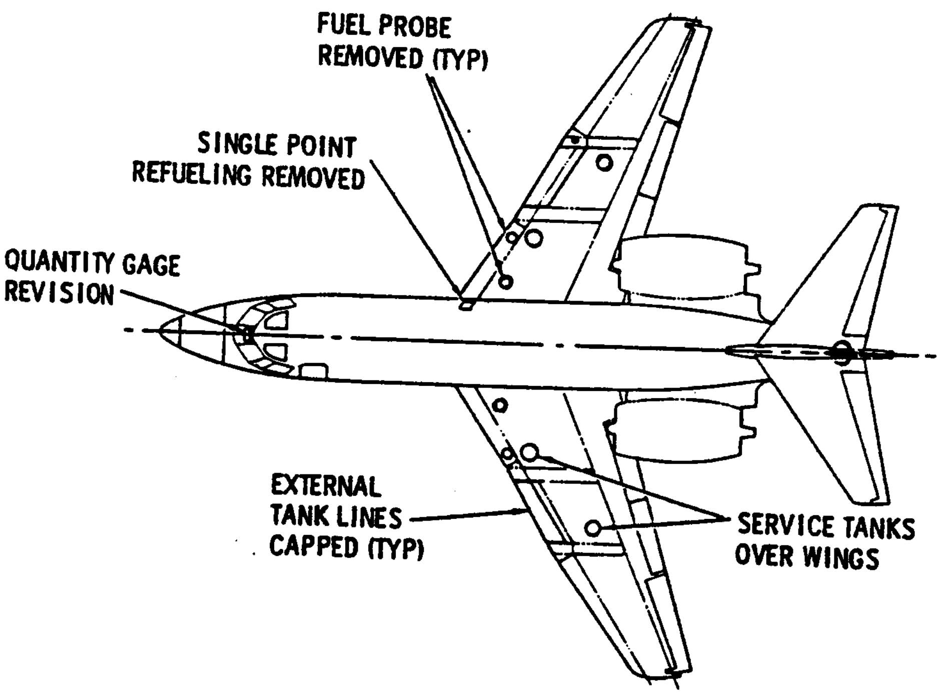

Figure 162. Fuel System Revisions

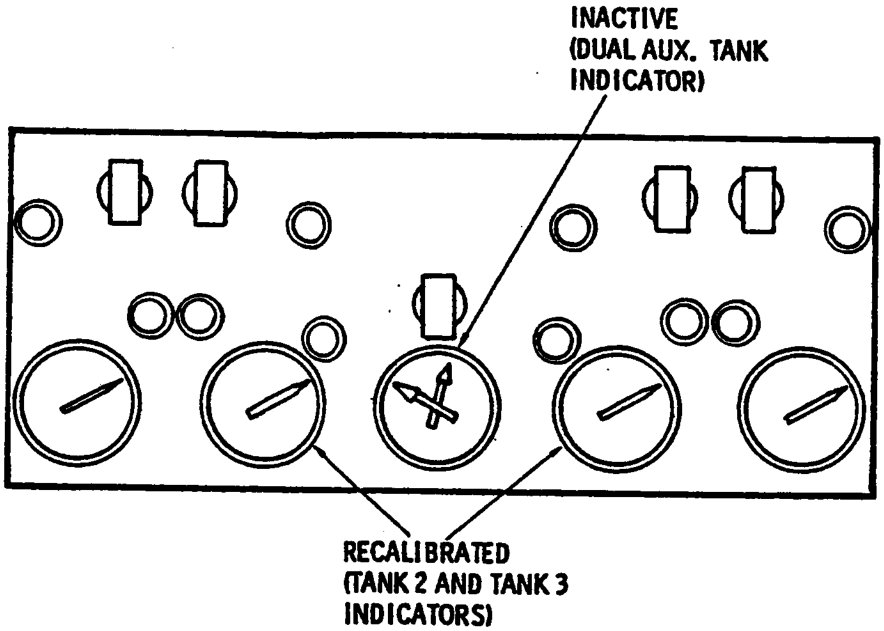

Figure 163. Fuel Quantity Indication

A fuel reduction of 3,700 kg (8,160 lb) results from the LFC changes. These reductions are summarized in Figure 164. Total fuel is reduced from 8,084 kg (17,822 lbs) to 4,383 kg (9,662 lbs).

6.1.6 Wing Ice Protection

Production configuration JetStar de-icer boots are installed on the wing fixed leading-edge sections and on the two outboard leading-edge flap sections, while a fluid anti-icing system is provided for the leading-edge test articles. This arrangement is shown in Figure 165.

Changes to the basic JetStar wing ice protection system are shown in Figure 166.

6.1.7 Hydraulic System

Both leading-edge and trailing-edge flaps on the JetStar may be operated from either the Number 1, Number 2, or emergency hydraulic systems, but the leading-edge flaps are inactivated on the LFC-LEFT modification. As both leading- and trailing-edge flaps are actuated from a single pilot's control, some revisions to the hydraulic system are required. This, as shown in Figure 167, consists of deactivating the leading-edge flap selector and capping of the hydraulic lines. The emergency hydraulic system does not show on the hydraulic schematic diagram, as it is an electric motor-driven system that backs-up the JetStar Number 1 system.

The hydraulic system for actuation of the McDonnell Douglas test-article shield is installed so shield actuation can be done by use of power from any JetStar system. The schematic for the added McDonnell Douglas shield hydraulic system is shown in Figure 168.

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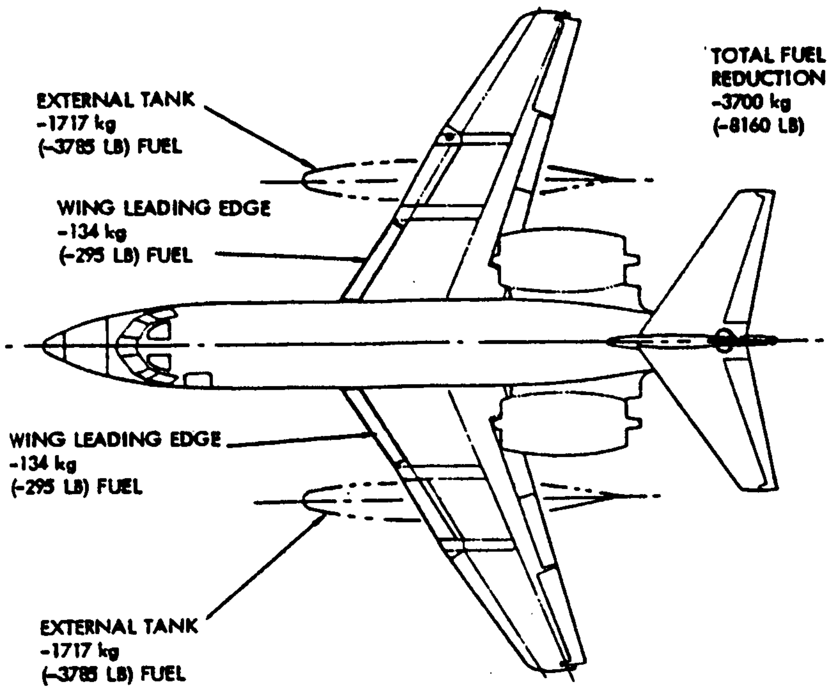

Figure 164. JetStar LEFT Fuel Reductions

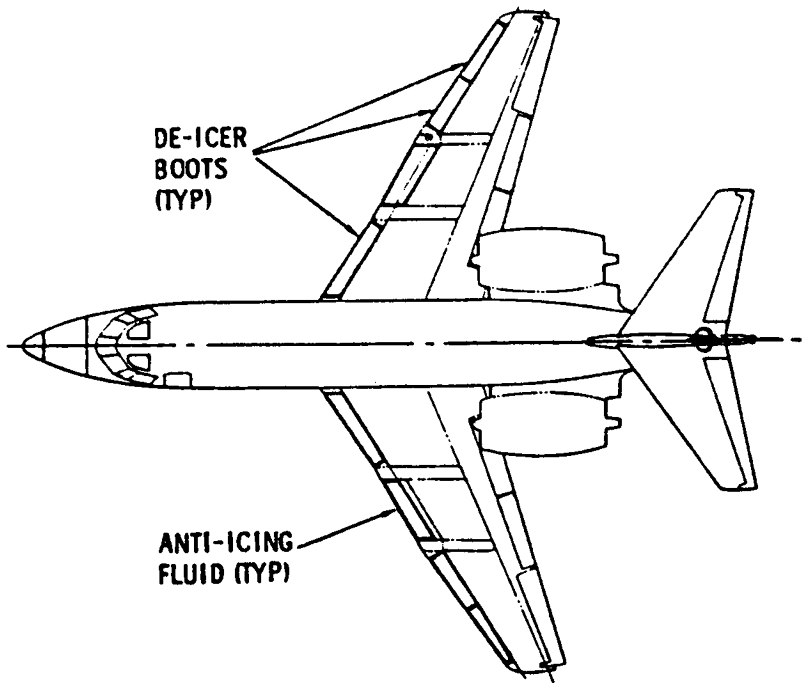

Figure 165. Wing Ice Protection

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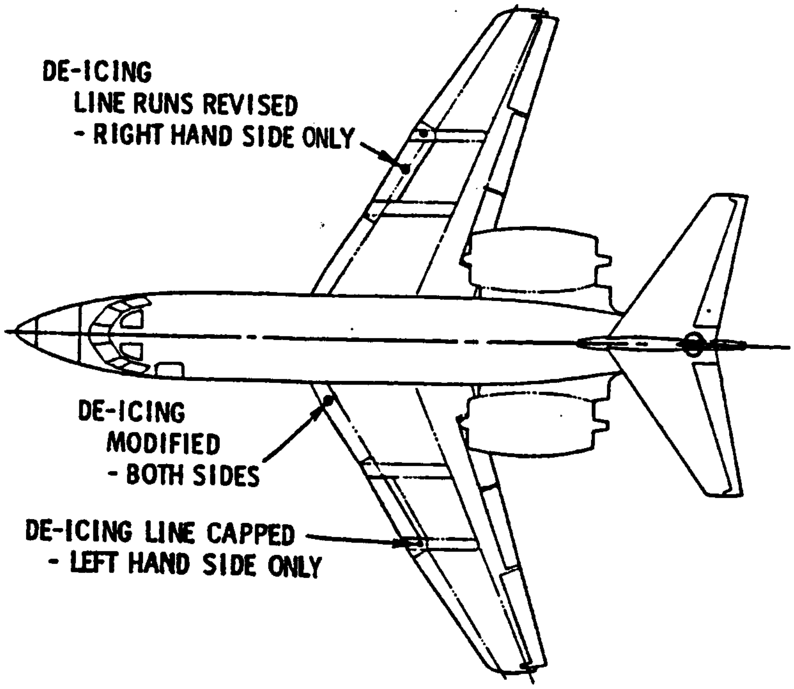

Figure 166. JetStar De-Icer System Changes

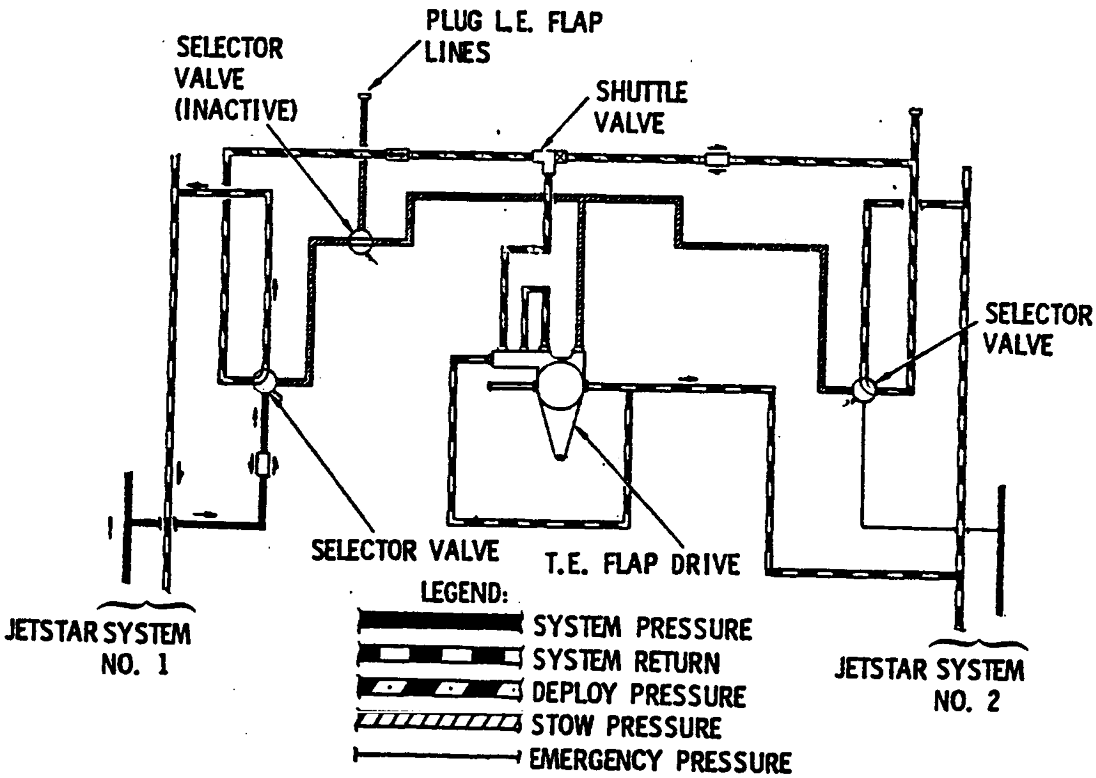

Figure 167. JetStar Hydraulic System Revision

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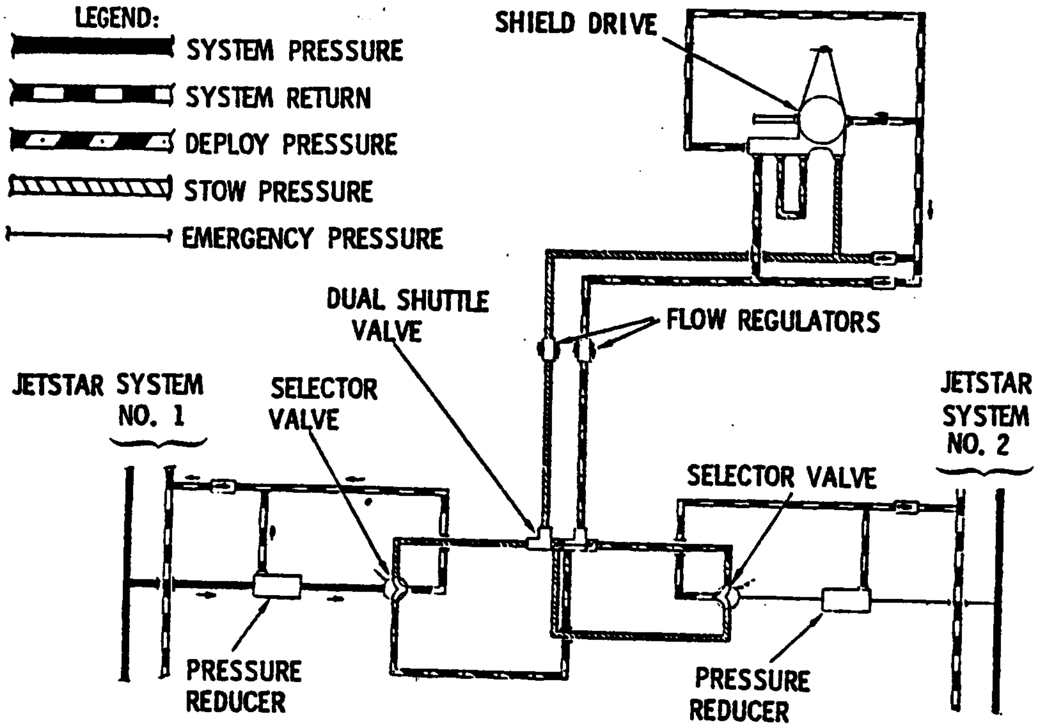

Figure 168. McDonnell Douglas Shield Hydraulic System Schematic

A hydraulic panel containing the dual shuttle valve, selector valves, and pressure-reducing valves for the shield hydraulic system is installed under the cabin center walkway floor at fuselage station 412. This panel and its location are shown in Figure 169.

6.2 LFC SYSTEMS

Numerous changes and hardware additions are required for the LFC experiments. Figure 170 shows the internal arrangement of the LFC JetStar. Suction lines from the LFC test sections are routed inboard through the JetStar fixed leading edges, through the fuselage walls, and into the flowmeters or flowmeter adapters installed on the seat deck in the cabin.

Suction lines run from the flowmeter position aft to the chamber valves, which are mounted vertically on support frames attached to existing seat rails. Suction lines from the top of the chamber valves are routed into the main suction line that runs aft through the pressure bulkhead to the suction pump mounted in the aft-fuselage compartment.

Cleaning-liquid tanks are also mounted in the aft compartment. A control console is installed on each side of the forward cabin area. A master suction pump switch and a McDonnell Douglas leading-edge shield switch are installed at the pilot's station.

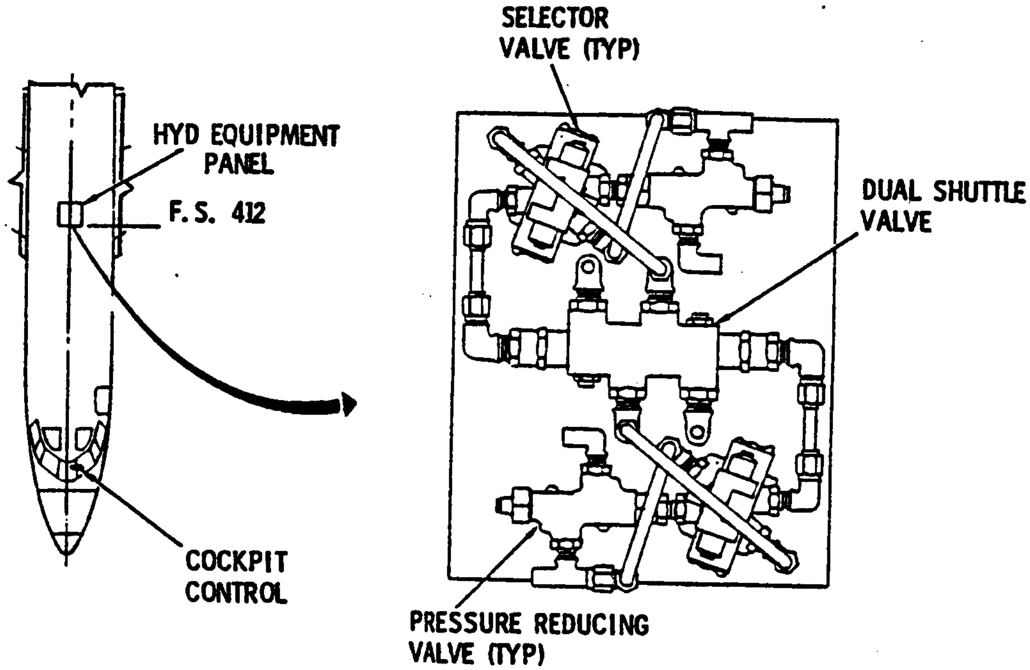

Figure 169. Shield Hydraulic Panel

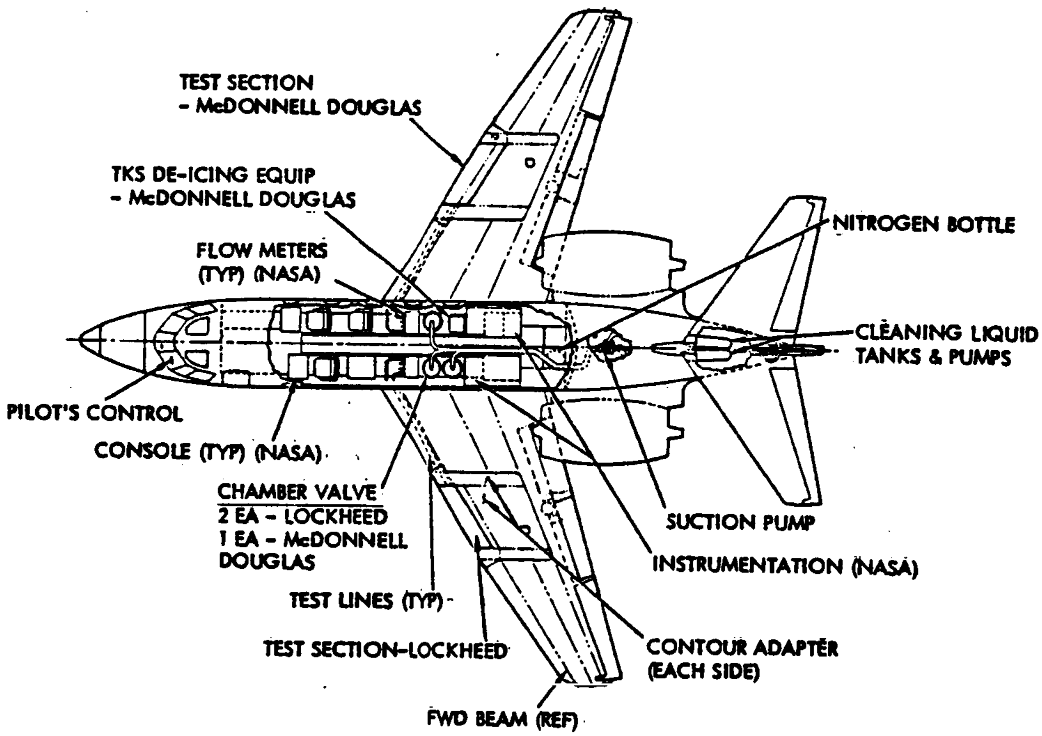

Figure 170. LFC-LEFT Systems Installation

6.2.1 Flowmeter and/or Adapters

Figure 171 shows flowmeter and flowmeter adapter locations on each side of the cabin just forward of the suction control chamber valves. Suction lines are routed from the wing leading edge through the fuselage wall and to the flowmeter position. NASA Langley determined which suction lines have flowmeters and also furnished the flowmeters and flowmeter line adapters. McDonnell Douglas suction lines with flowmeter installed are lines 1, 3 and 5.

Lockheed suction lines with flowmeters installed are U1, U2 and D2. Flowmeters may be switched to other lines if necessary to check flow in those lines. Development and installation of the suction lines in these areas was the responsibility of NASA Dryden.

6.2.2 Chamber Valve Installation

Three chamber valves are installed in the JetStar cabin, as shown in Figure 172. One valve is installed on the right-hand side of the cabin for the McDonnell Douglas suction lines, and two chamber valves are installed on the left-hand side for the Lockheed suction lines. The valves are mounted vertically with the suction lines from the wing entering from below; the main suction lines exit from the top of the chamber valve.

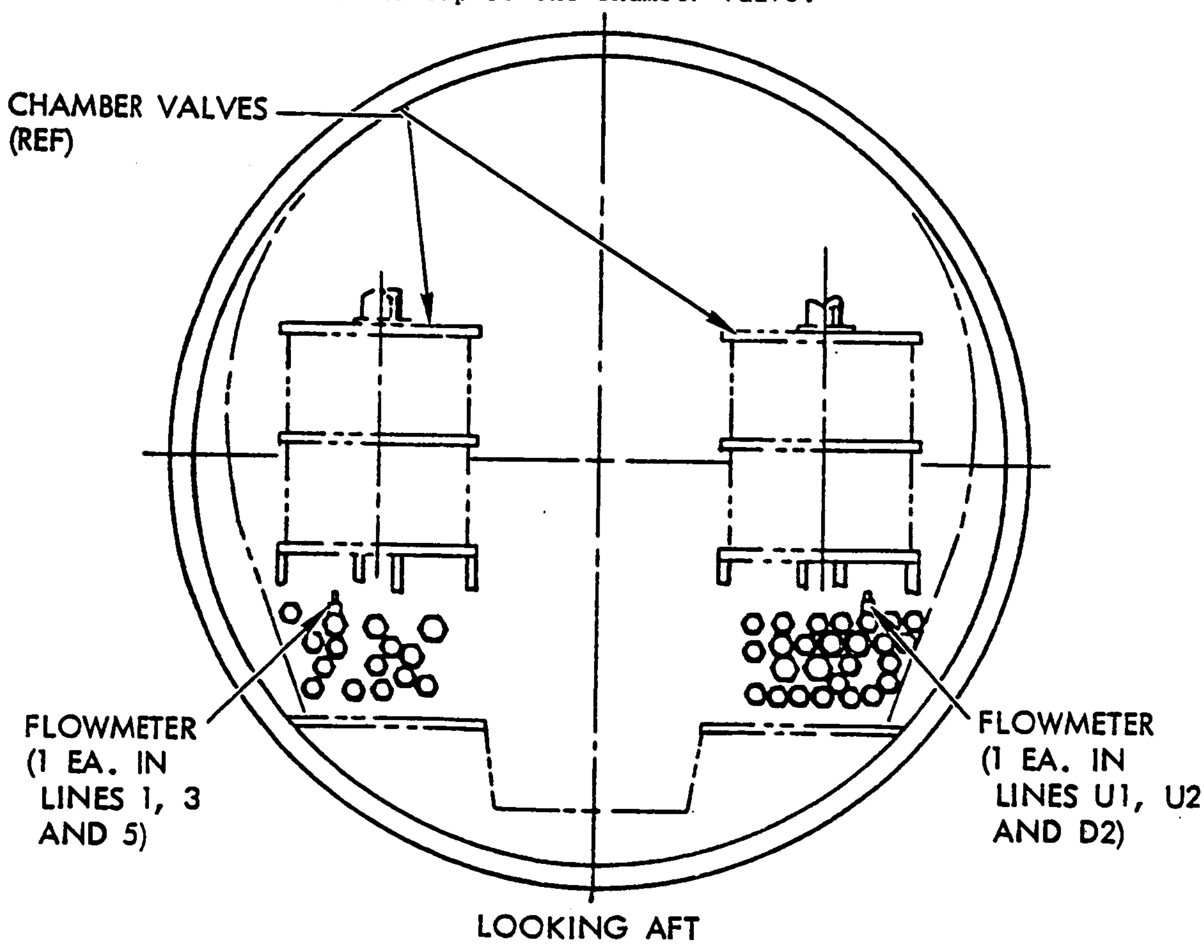

Figure 171. Flowmeter/Adapter Installation

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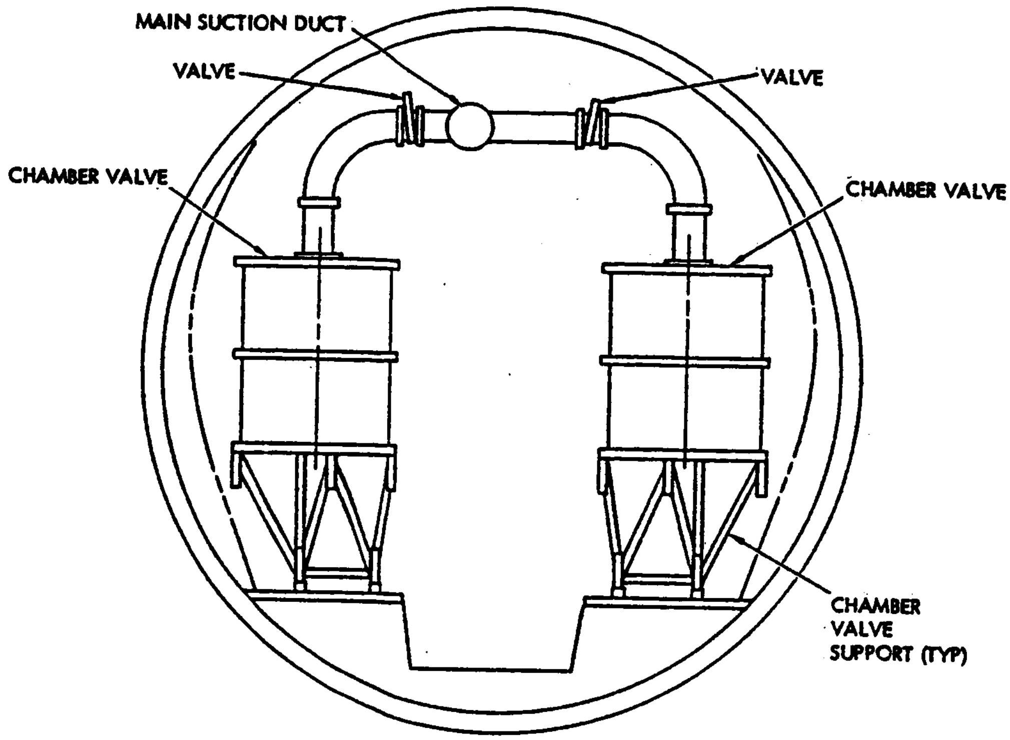

Figure 172. Chamber Valve Installation

Figure 173 shows a typical chamber valve support frame. The welded steel frame is attached to the seat tracks installed in the JetStar seat deck. The chamber valve is attached to the frame with bolts through its lower flanges.

6.2.3 Main Suction Line

Suction lines from each chamber valve are manifolded into the main suction duct, which is routed along the top of the aft cabin through the JetStar aft pressure bulkhead to the suction pump installed in the aft fuselage compartment.

A shutoff valve is provided for the main line and for each chamber valve as shown in Figure 174.

6.2.4 Suction Pump Installation

The LFC suction pump is a compressor unit modified by Garrett Corporation (AiResearch) for this application. The pump is mounted in the aft fuselage compartment of the JetStar as shown in Figure 175, in the approximate location normally reserved for an auxiliary power unit installation. Although blade containment is adequate within the pump case, additional external shielding is provided for hub containment. The existing JetStar engine bleed cross-manifold supplies power to the suction pump turbine.

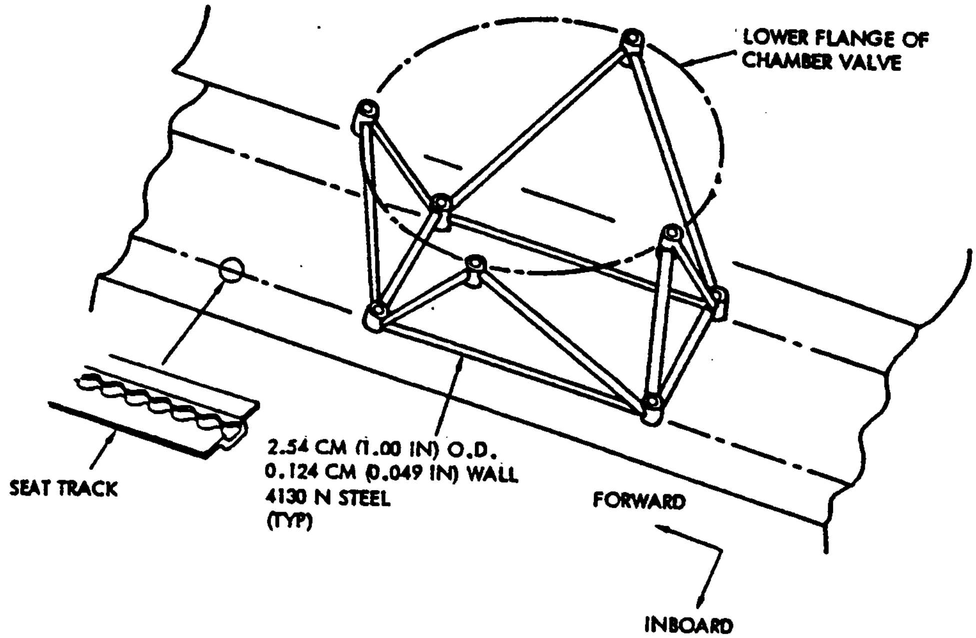

Figure 173. Chamber Valve Support

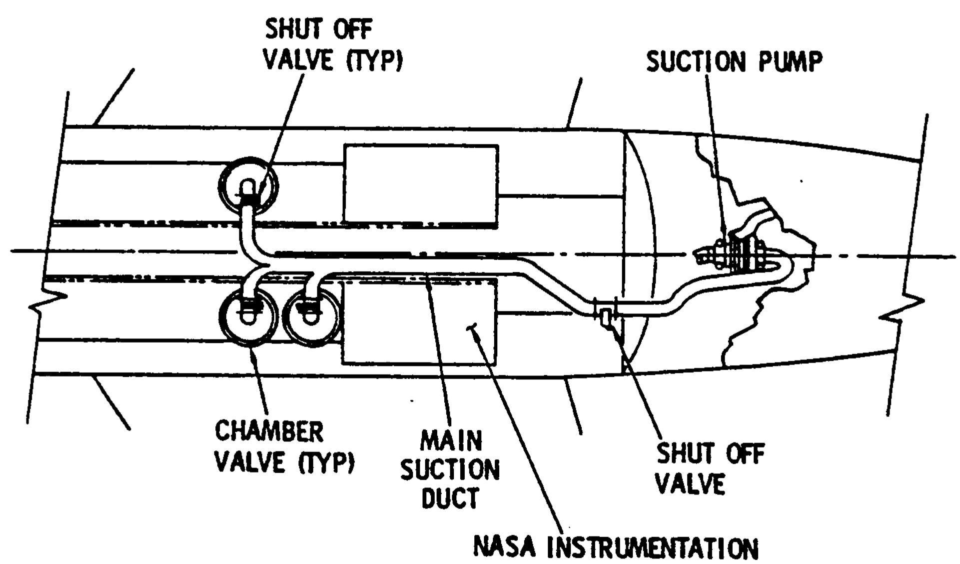

Figure 174. Main Suction Duct Installation.

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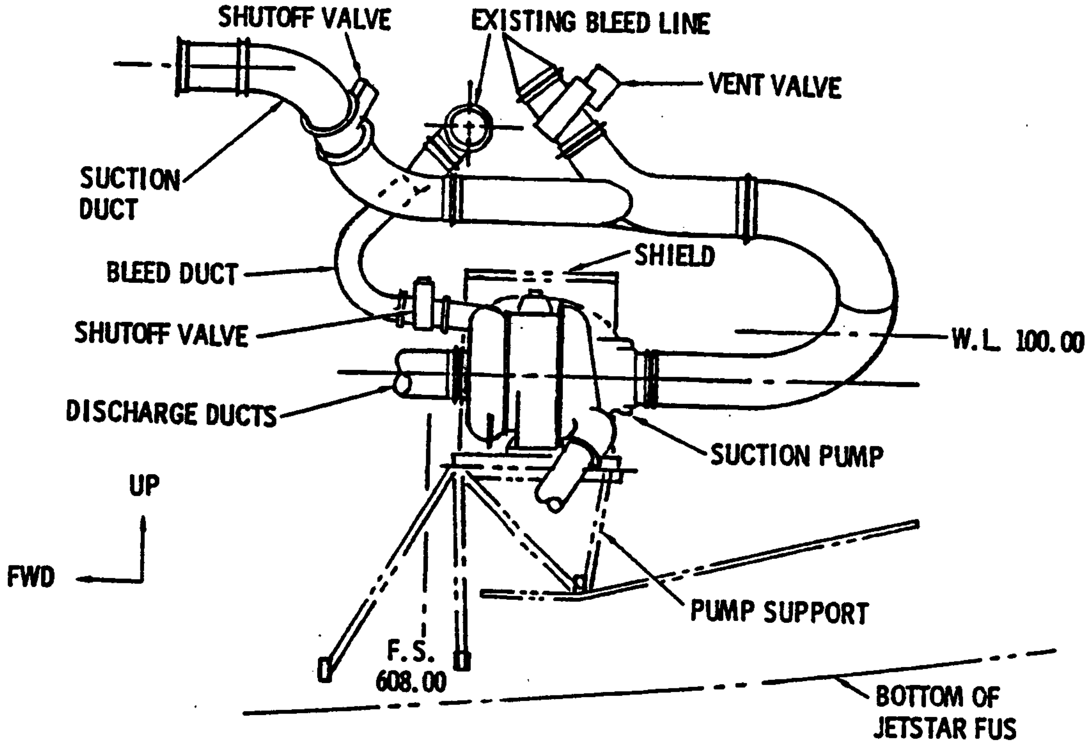

Figure 175. Suction Pump Installation

Discharge ducts from the turbine and compressor outlets are vented overboard. Venting of the suction line is provided to prevent pump compressor stall.

To limit the temperature of the self-contained turbo-compressor oil supply, a cooler is installed as shown in Figure 176. This cooler is made by packaging three Kool-Mar, Dunham-Bush, Inc., Part Number 10318-R1-0338 heat exchangers in series with required cooling air ducting. Control of oil flow through the cooler is provided by a temperature control valve installed within the oil cooler package.

6.2.5 Purge System

Both leading-edge test articles require purging of the suction systems to prevent collection of anti-icing/cleaning fluids or water. Purge air is provided from the basic JetStar systems. The primary purge system, shown in Figure 177, supplies pressurized air from the JetStar emergency pressurization system. The emergency pressurization system is tapped just downstream of the heat exchanger and ducted through an air filter into the main suction duct. An emergency pressurization system diverter valve and a primary purge system shutoff valve are installed to control the LFC purge operation. Use of the primary purge system is prohibited below 3,658m (12,000 ft) altitude due to excess purge-air temperature and aft-compartment temperature below this altitude.

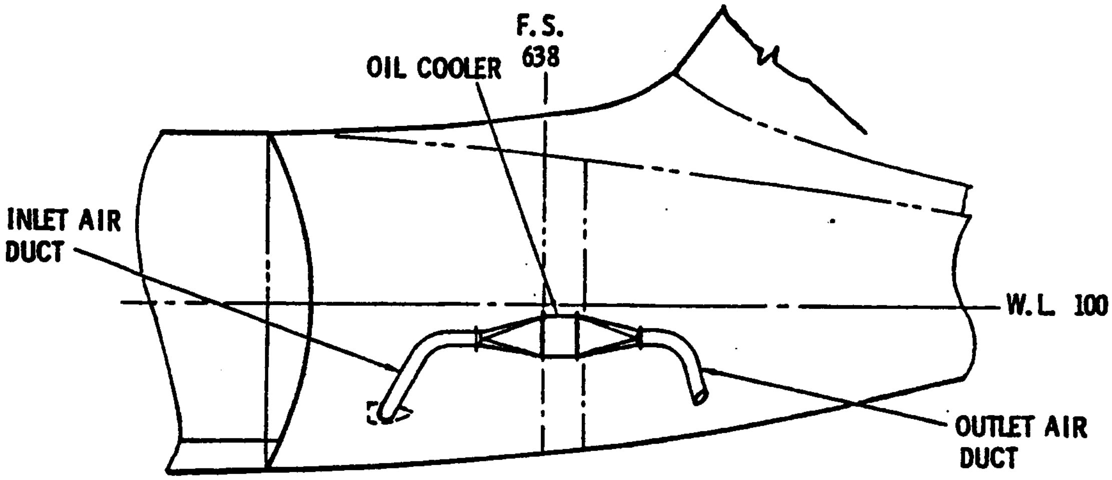

Figure 176. Oil Cooler Installation

A secondary system, shown in Figure 178, provides purge air for use below 3,658 m (12,000 ft), for the McDonnell Douglas LFC system. Secondary purge air is obtained by diverting air from the primary air-conditioning supply duct. A diverter valve and secondary purge system shut-off valve are installed for system control. The secondary purge system is manifolded with the primary system just before the primary system duct enters the purge-air filter.

Openings are made in the JetStar aft pressure bulkhed to accommodate the purge systems ducting as shown in Figure 179.

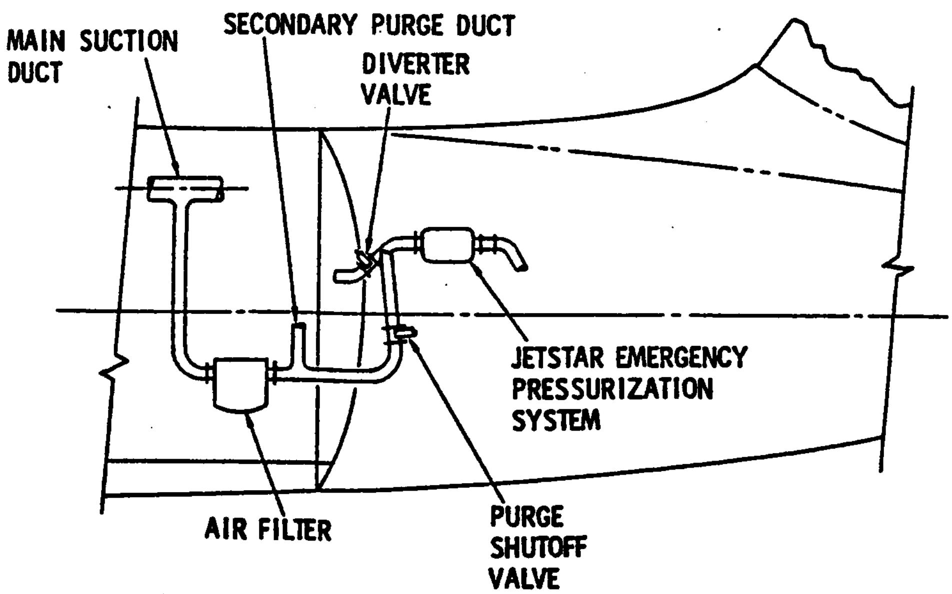

Figure 177. Primary Purge System

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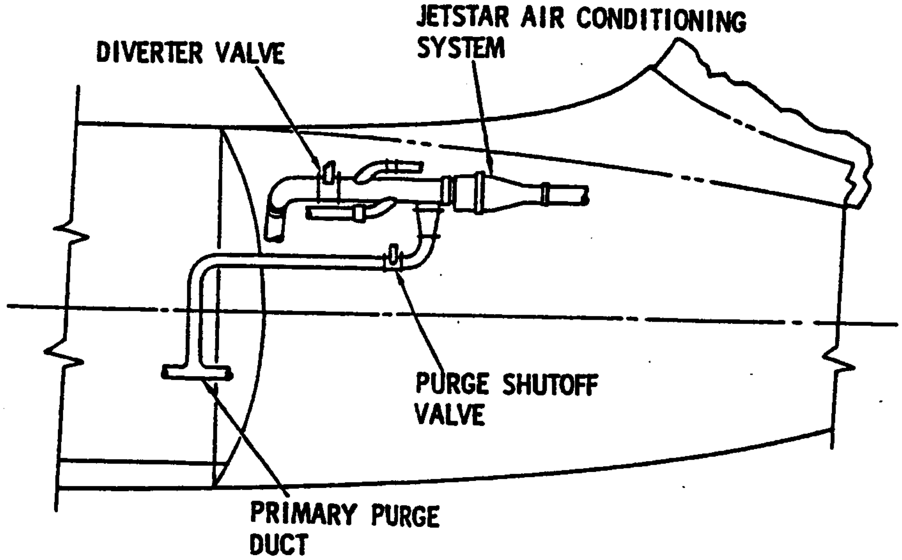

Figure 178. Secondary Purge System

6.2.6 Cleaning/Anti-Icing Fluid Tank Installation

Two cleaning/anti-icing fluid tanks are installed in the lower portion of the JetStar aft fuselage. Location of these 60.94 liter (16.00 gal)tanks is shown in Figure 180. The right tank is for the McDonnell Douglas LFC system, and the left tank is for the Lockheed LFC system.

Intercostals and saddle assemblies installed between existing JetStar fuselage frames provide support for the fluid tanks as shown in Figure 181.

Control panels, shown in Figure 182, for the cleaning/anti-icing systems are installed in the aft fuselage just forward of the fluid tanks aft of the speed brake. Provisions for tank filling, draining, and nitrogen pressurization are incorporated within the control panels. Fluid lines run from the control

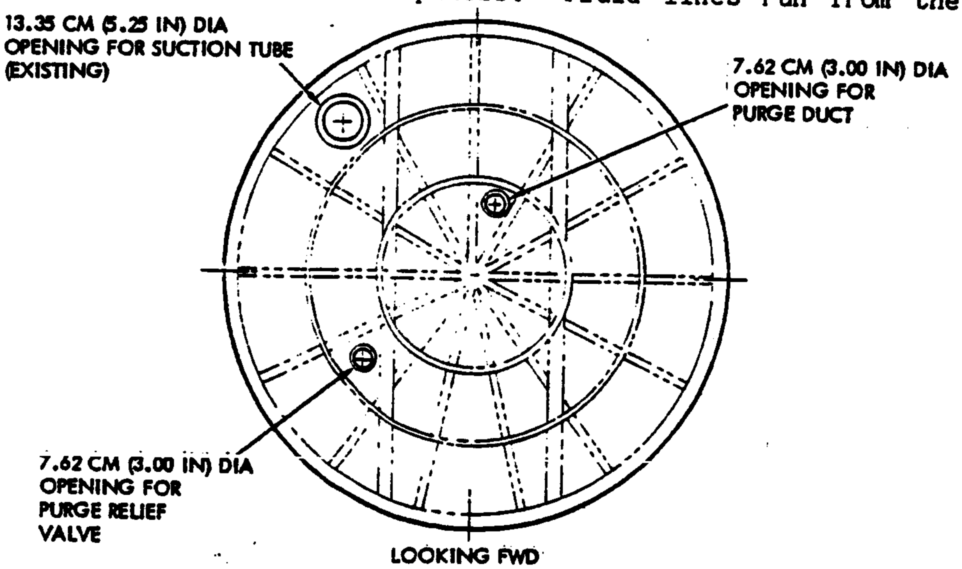

Figure 179. Pressure Bulkhead Openings

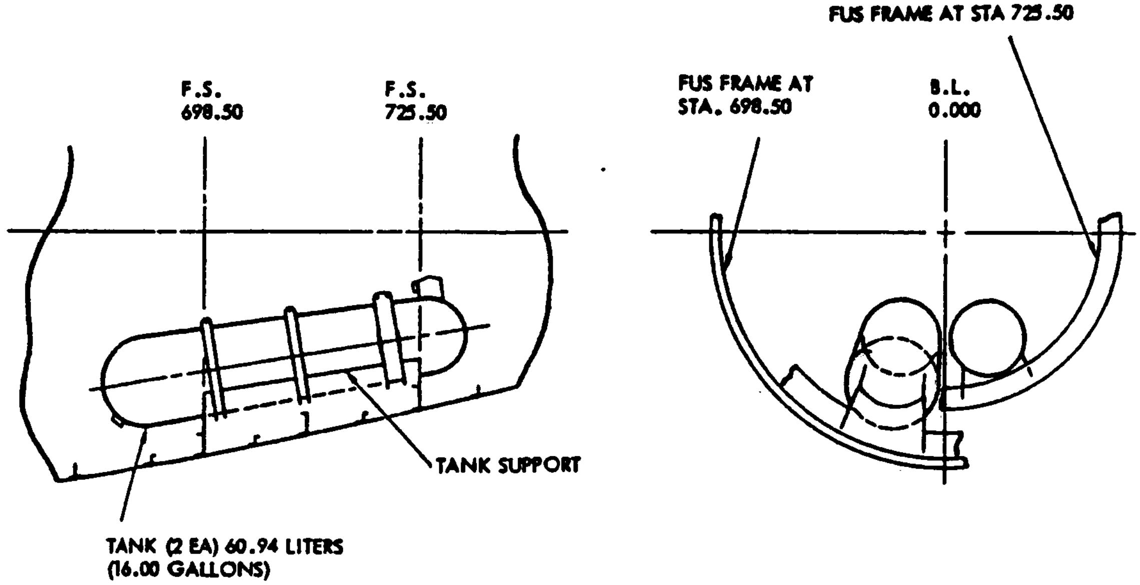

Figure 180. Fluid Tanks Installation

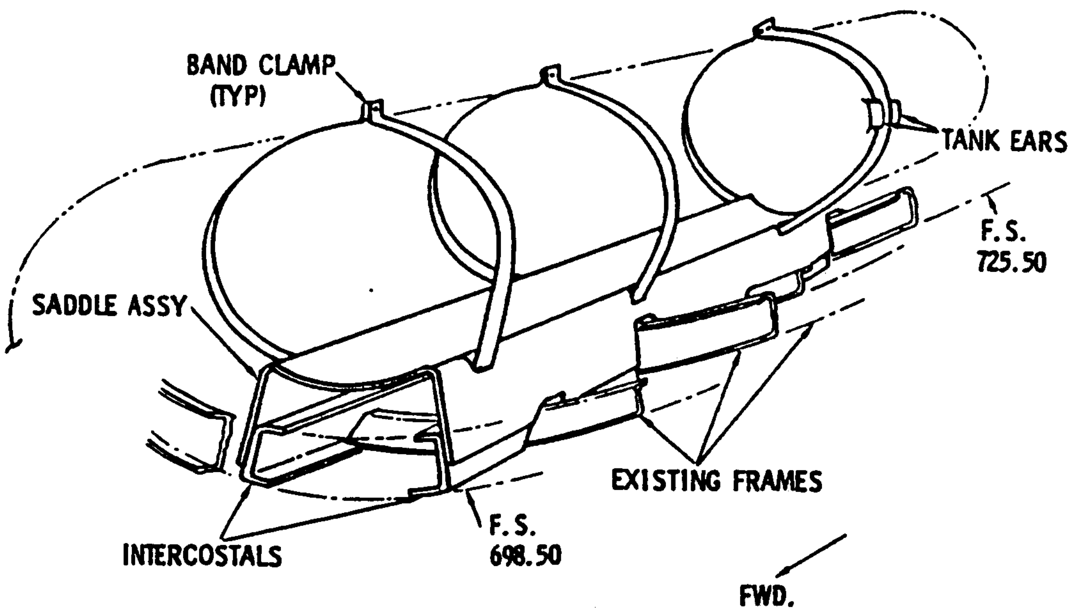

Figure 181. Fluid Tank Support

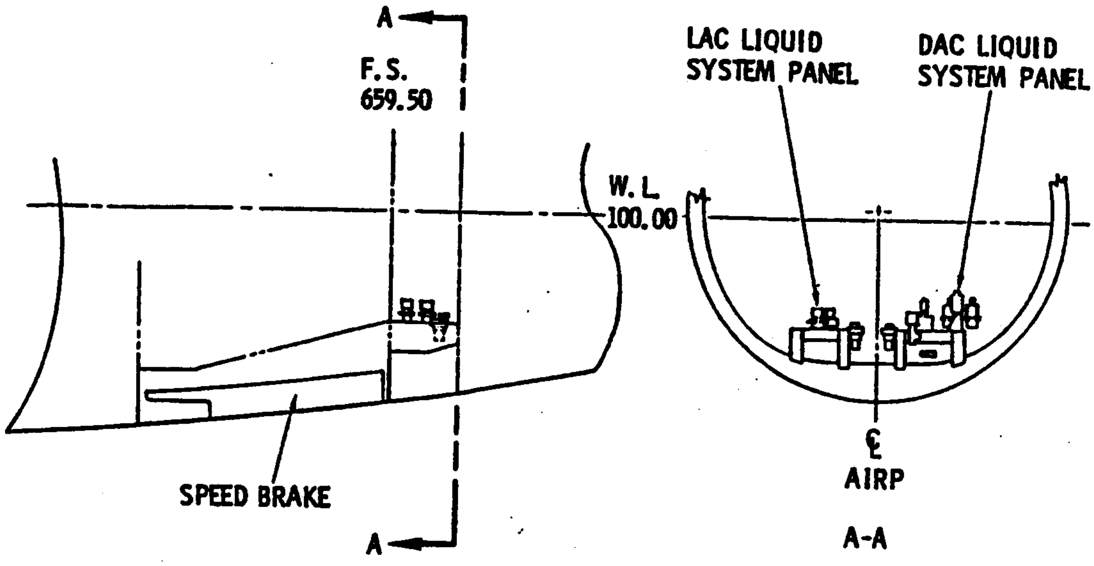

Figure 182. Fluid System Control Panels Location

panel through the JetStar wing fillets to the test articles. No cleaning/anti-icing fluid is routed through the aircraft cabin.

As described in Section 5.1, some of the slots on the Lockheed test articles serve a dual purpose: for fluid distribution and for suction. To avoid running extra tubes through the JetStar inboard wing leading edge, it is necessary for these tubes to also serve a dual purpose. This is accomplished by installing remotely controlled 3-way valves in the left wing-to-fuselage fairing, as shown in Figure 183.

6.2.7 Aft Fuselage Openings

Several openings are required in the JetStar aft fuselage skin to accommodate the LFC equipment installed in the aft compartment. Turbine and compressor exhausts are vented overboard, and liquid systems service and bleed lines protrude through the fuselage skin. Figure 184 shows the approximate location of these overboard lines.

6.2.8 McDonnell Douglas Shield Anti-Icing

The McDonnell Douglas LFC system requires an ice protection system for the leading edge of the contamination avoidance shield. The liquid tank, pump, filter, and valves are mounted on a special rack installed on the right-hand cabin seat deck. This system was designed by TKS LTD and furnished by McDonnell Douglas; installation design was provided by Lockheed. The anti-icing mounting rack is shown in Figure 185, and its installation is shown in Figure 186.

6.2.9 Nitrogen System Container

A 40.64 cm (16.00 in) diameter spherical nitrogen container is installed on the left-hand side of the cabin just forward of the aft pressure bulkhead. The nitrogen pressure is used to purge instrument lines and to pressurize the

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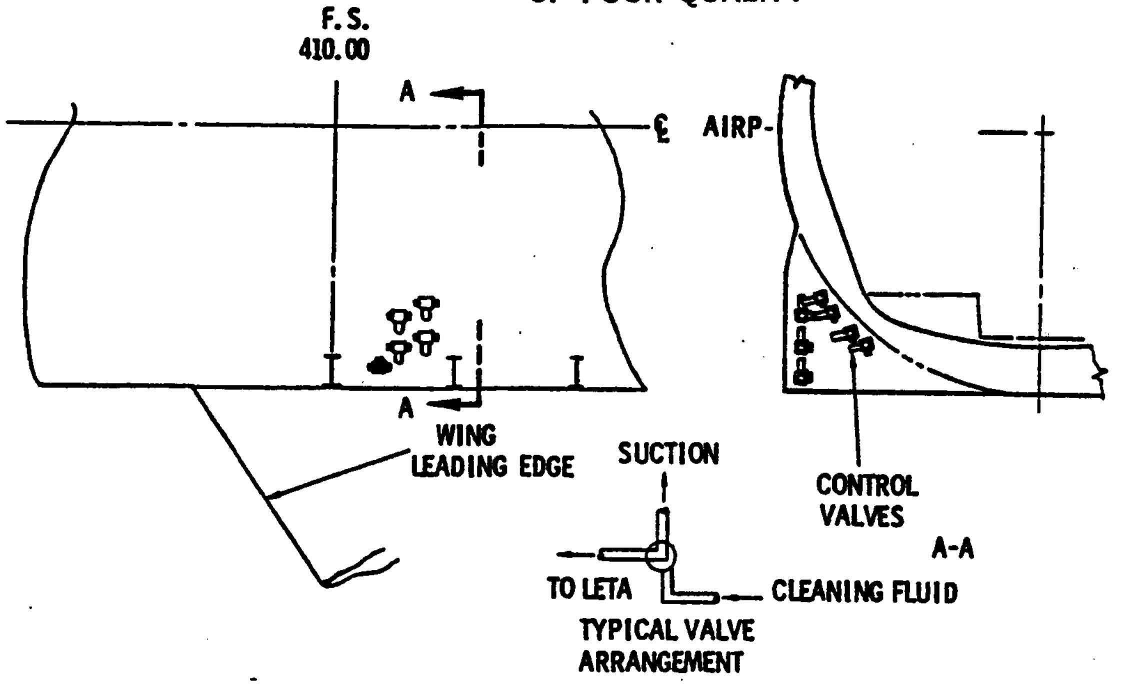

Figure 183. Suction/Fluid Selector Valves

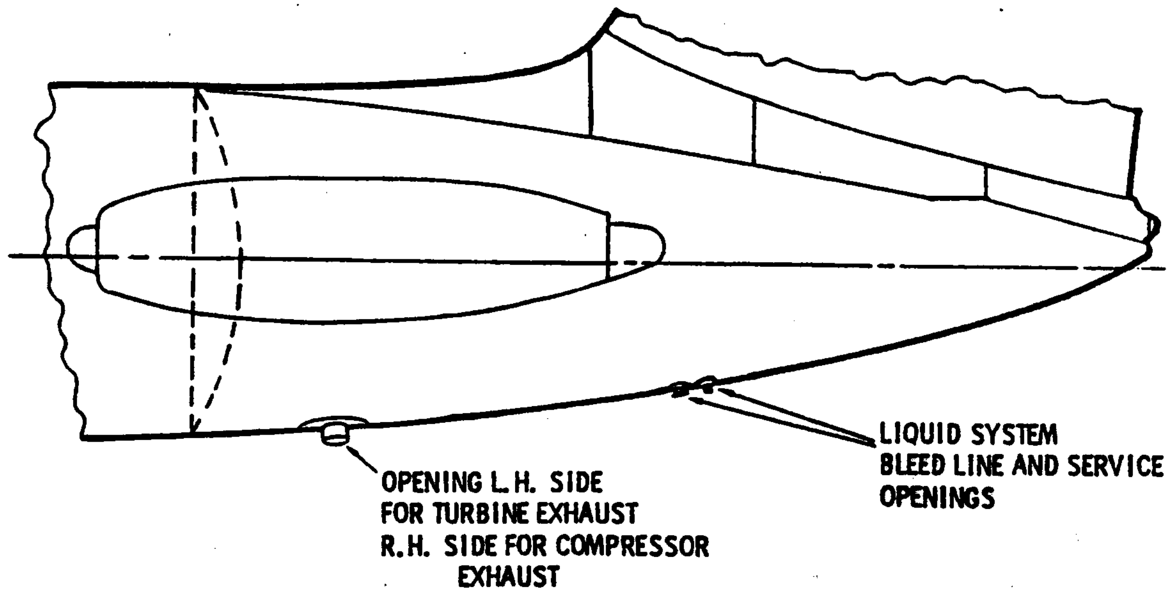

Figure 184. JetStar Aft Fuselage Openings

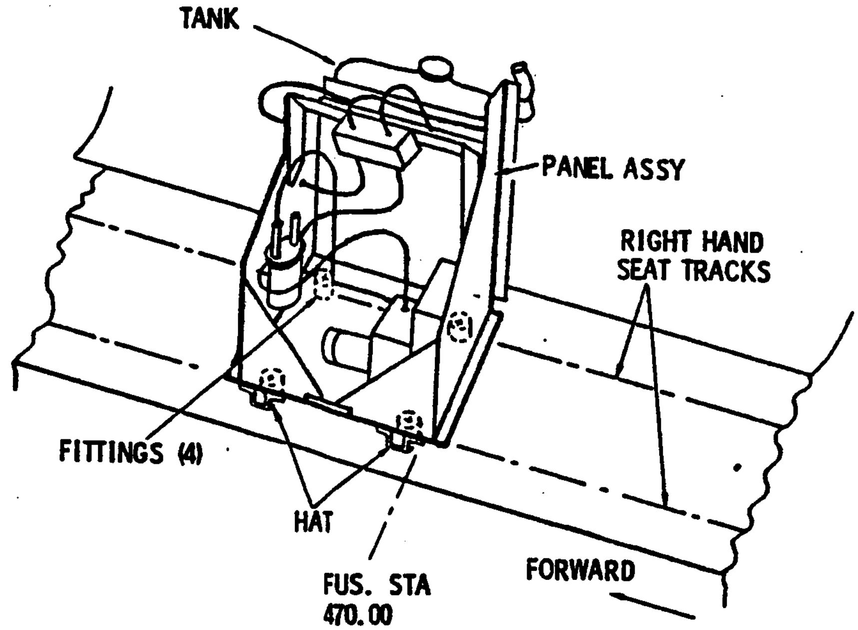

Figure 185. McDonnell Douglas Shield Anti-Icing Rack

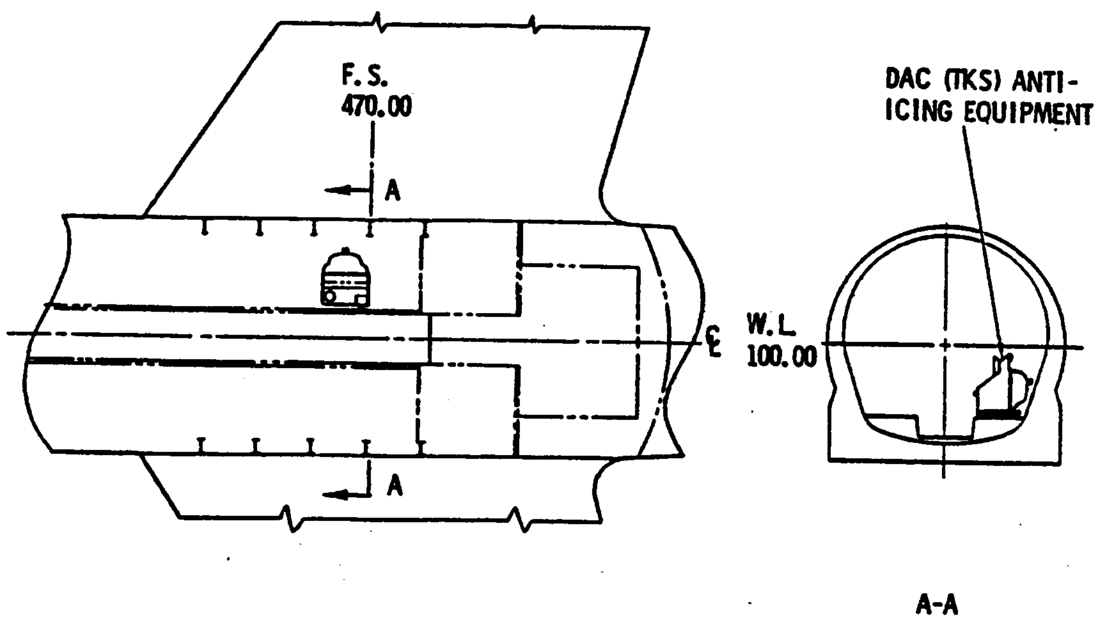

Figure 186. Anti-Icing Rack Location

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cleaning-fluid tanks installed in the JetStar aft compartment, and to purge the McDonnell Douglas contamination avoidance spray nozzles. Figure 187 shows the location of the nitrogen container.

6.2.10 Instrument Inverter Installation

Some of the LFC-LEFT instruments installed by the NASA DFRF requires 115-volt, 60-cycle electrical power, which is not available on the basic JetStar. A NASA DFRF selected and installed inverter is located in the JetStar cabin just across from the nitrogen container to provide the electrical power required. Figure 188 shows the location of this installation.

6.2.11 Control Consoles

Control consoles designed by the NASA DFRF are installed one on each side of the forward cabin. The left-hand console is for the Lockheed LFC system, the right-hand console for the McDonnell Douglas LFC system. Suction pump controls are mounted next to the CRT in the McDonnell Douglas console. The switch for the main suction line shut-off valve is located above the CRT on the upper console panel. Figure 189 shows this arrangement.

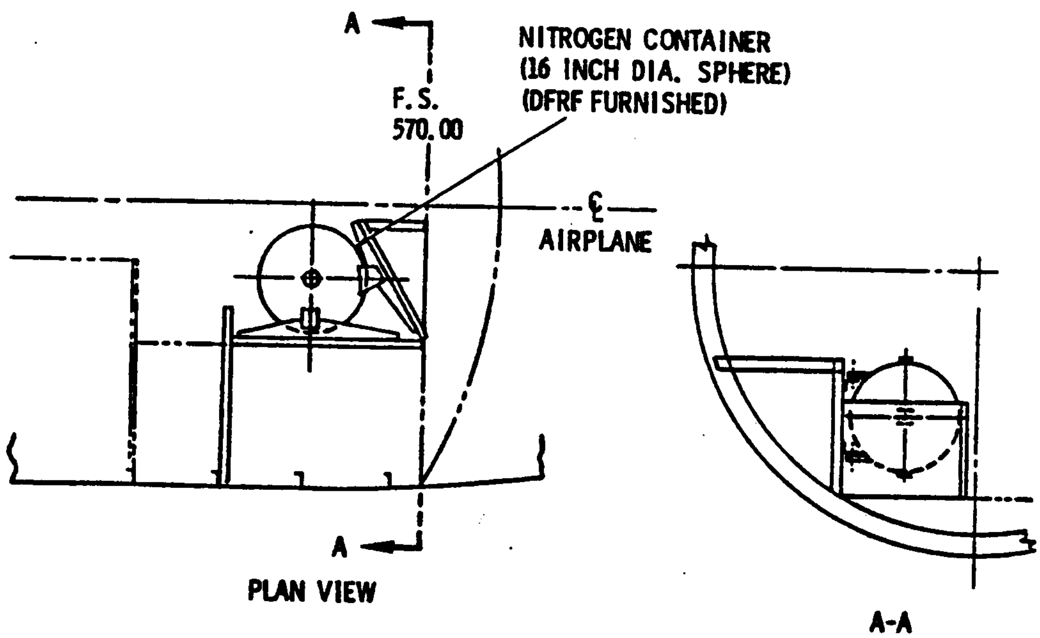

Figure 187. Nitrogen Container Installation

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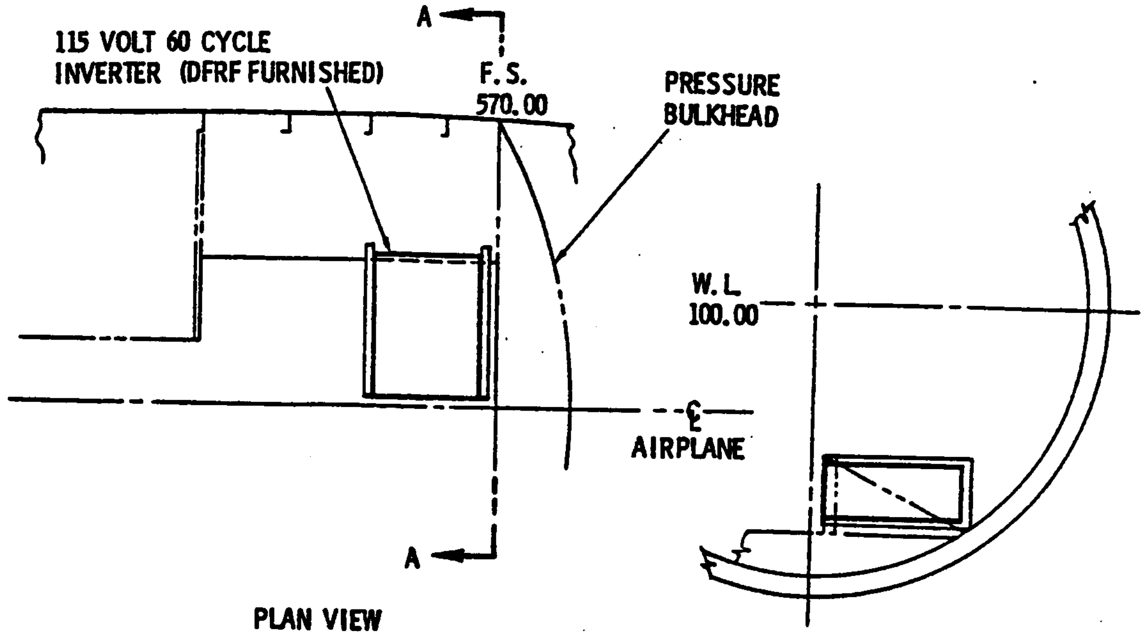

Figure 188. Instrument Inverter Installation

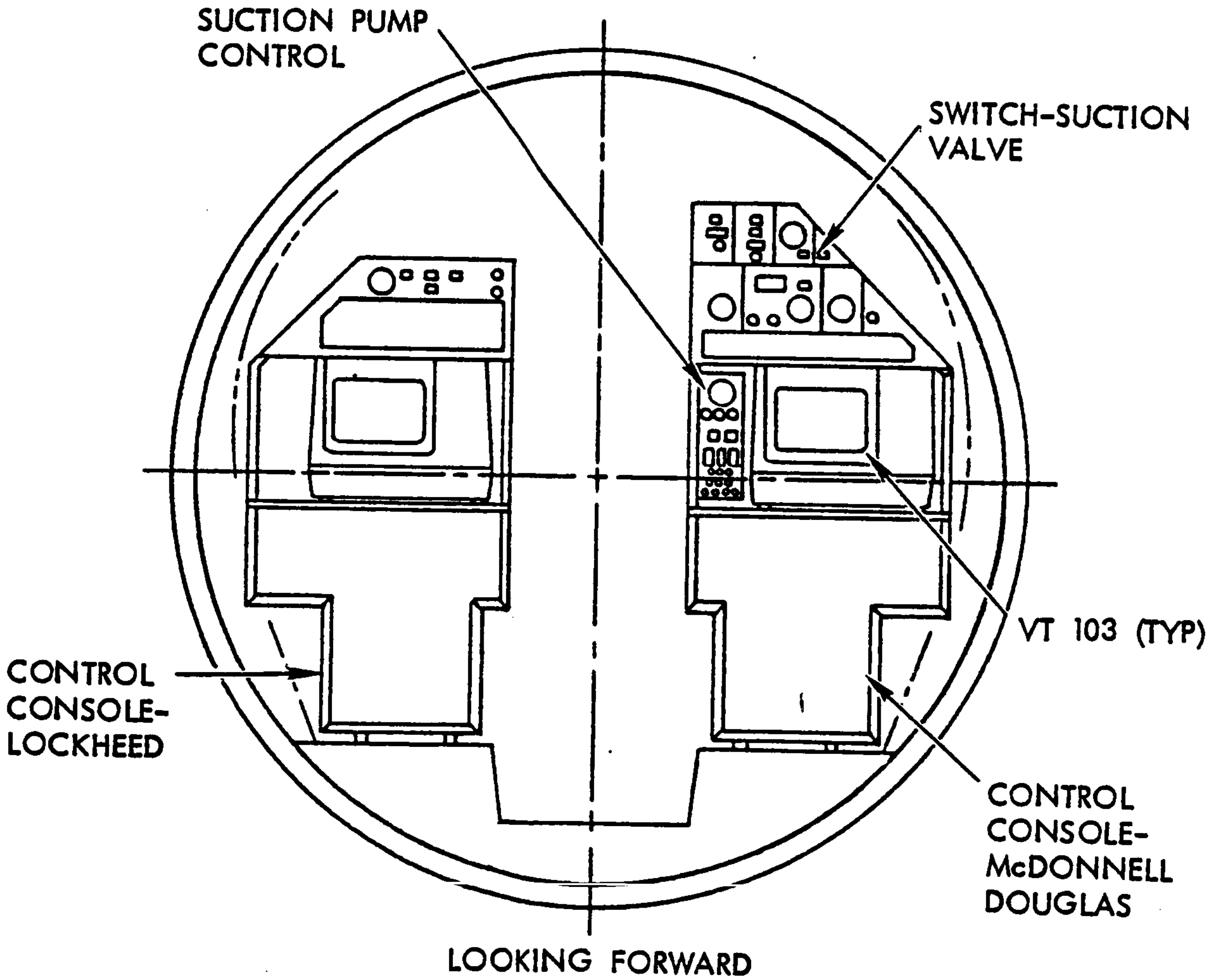

Figure 189. LFC-LEFT Control Consoles

7.0 AERODYNAMICS AND PERFORMANCE

This section covers five general topics:

(1) The design philosophy of the glove modifications prior to the Calspan wind tunnel test.

(2) The equivalent wing methodology, which forms the basis for the theoretical pressure calculations.

(3) Results of the Calspan wind tunnel test and correlation of experimental and theoretical pressure data.

(4) Pressure distributions and boundary layer results for the final glove contour.

(5) Aircraft take-off and landing performance.

7.1 EARLY GLOVE MODIFICATIONS

Several changes in design philosophy occurred during the glove design evolution. Design point changes necessitated major changes to the glove contours to achieve the desired pressure distribution. This section summarizes the contour modifications prior to the Calspan wind tunnel test. Figures 190 and 191 show pertinent contour shapes and the resulting pressure distributions for an outboard glove station, where it was most difficult to achieve the desired pressure distribution. The difficulty arises from the strong three-dimensional effects inherent when placing a limited span glove on an existing wing.

7.1.1 First Glove Contour (MOD 1)

The glove pressure distributions were intended to be representative of those on a 1990's LFC aircraft wing. Accordingly, the initial glove design point was M = 0.80 at 12,192 m (40,000 ft). Referring to Figure 190, the first contour (MOD1) was developed for the JetStar feasibility study performed under NASA contract NAS1-14631 in February 1979. The pressure distributions were calculated by the Bailey-Ballhaus Transonic Wing Program (Reference 1). At the outboard station, the pressure distribution was not very satisfactory. Changes were made to the outboard station contour to fill out the pressure distribution, and this glove was designated MOD2. A three-dimensional analysis of this glove was never made, but the two-dimensional pressure distribution was factored by the local sweep angles to give a representative three-dimensional distribution. As can be seen in Figure 190, the outboard modification did result in filling out the pressure distribution.

7.1.2 MOD5B Contour

The MOD2 glove was close to an acceptable contour, and a full three-dimensional analysis would have been made, but it was at this time that the design

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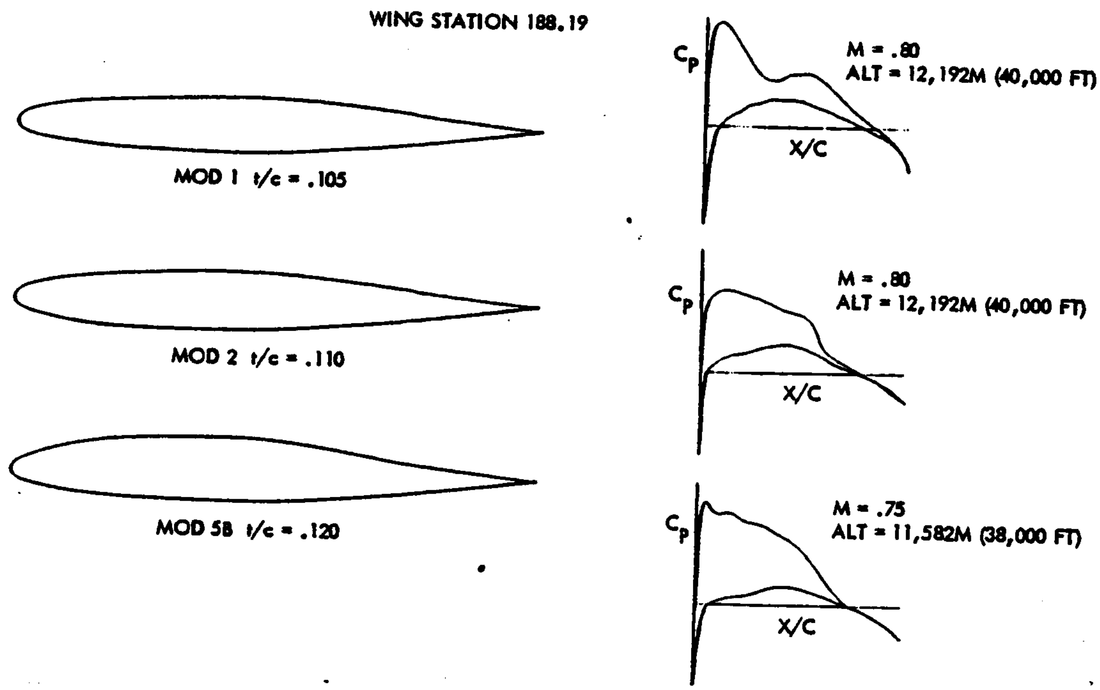

Figure 190. Early Glove Contour—MOD1, MOD2, MOD5B

point began to change. Contours designated MOD3 and MOD4 were the result of brief studies at M = 0.775, before the existing design point of M = 0.75 at 11,582 m (38,000 ft) was established. MOD5B was established at this design point. Additionally, MOD5B pressures were to result in a local Mach number of 1.10 at 5 percent chord, and 1.02 at 40 percent chord. That goal was to become important again during the Calspan wind tunnel test. Due to the change in design point, the MOD5B contour was 1 percent thicker than the previous contours. This glove contour was the first to have the three-dimensional analysis done with the FLO22NM code. All subsequent gloves were analyzed using this code.

7.1.3 DAC4 Contour

This MOD5B glove was close enough to a glove being developed independently at McDonnell Douglas that Lockheed personnel coordinated with McDonnell Douglas to determine a common contour. The DAC4 modification, shown in Figure 191, was selected from this work. This design had a pressure distribution that was acceptable to all parties, although it tended toward a "rooftop-type" distribution. This tendency was more obvious at stations further inboard.

7.1.4 MOD7Q Contour

Some slight changes were made to DAC4 by Lockheed at the outboard station to extend the rooftop further aft, which resulted in glove MOD7A. This was an acceptable shape and pressure distribution. However, when boundary layer calculations and suction requirements were made, it was found that an unacceptably large spanwise pressure gradient existed on the glove. This led to MOD7Q, which was the contour specified for the wind tunnel model.

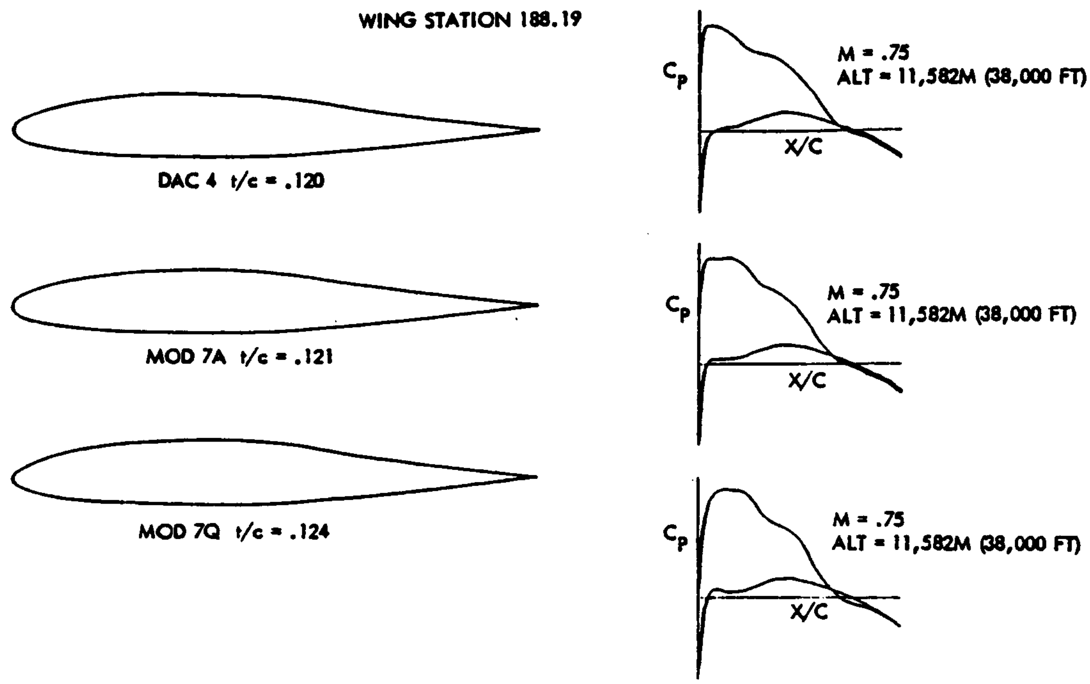

Figure 191. Early Glove Contours, DAC4, MOD7A, MOD7Q

7.2 EQUIVALENT WING METHOD

The calculated pressure distributions were based on an equivalent wing geometry. As used in this report, an equivalent wing is one which has been altered such that, when run on an isolated-wing computer code, the pressures calculated account for the interference effect of the body, pylons, and nacelles. FL022NM was the isolated-wing code used for all analyses from MOD5 to the final MOD8C glove.

7.2.1 Hess Code Method

The method for calculating the equivalent wing uses the Hess three-dimensional full-potential flow code (Reference 2) to evaluate the interference effects of the body, pylons and nacelles on the wing. Figure 192 depicts the wing/body/pylon/nacelle panel model used in the Hess code. It is a very accurate representation of the JetStar. The process for defining the equivalent wing, shown schematically in Figure 193, is:

(1) Run the Hess code for the wing/body/pylon/nacelle configuration to calculate section pressure distributions.

(2) Each section pressure distribution is input to Lockheed two-dimensional inverse codes to calculate a theoretical section contour and twist.

(3) The process is repeated for a wing-alone configuration.

(4) The difference in the theoretical contours at each section defines a shape distortion, which is then added to the respective original sec-

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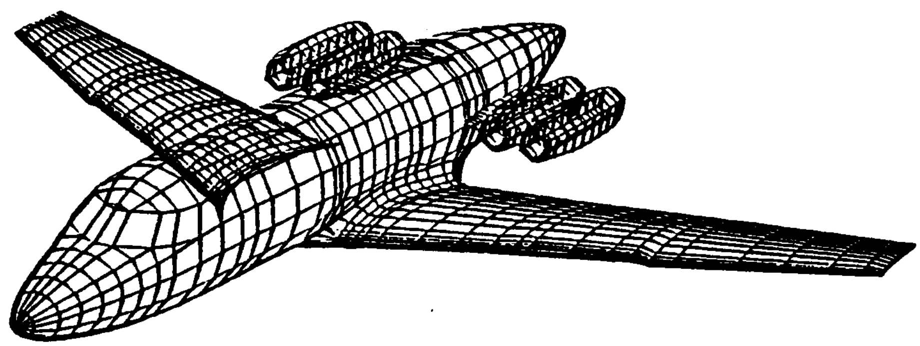

Figure 192. Hess Panel Model

tion contour to define an equivalent section that accounts for the interference effects of the body/pylon/nacelle.

(5) The equivalent wing is run on an isolated wing code (FLO22NM) to produce pressure distributions that account for the body/pylon/nacelle effect.

Figure 194 shows the section contours defined in the equivalent wing calculation process. The top contour represents a typical JetStar section contour. The middle two contours are the theoretical section contours calculated from the Hess pressures for the full configuration and the wing-alone runs. The difference in contours of these two sections, plus a twist change, when added to the original JetStar section, produces an equivalent wing section. This section is generally thicker than the original section, but still maintains a reasonable contour.

7.2.2 Hess Code Validity

The validity of this method is shown by correlations with theoretically calculated pressures and experimental pressures. Figure 195 compares pressure distributions calculated by the Hess code for a wing/body/pylon/nacelle configurations, a wing-alone case, and an equivalent wing-alone case. The equivalent wing pressures match the wing/body/pylon/nacelle pressures over most of the chord length. The correlation with wind tunnel data is presented in Section 7.4.

7.3 CALSPAN WIND TUNNEL TEST RESULTS

A 0.10-scale model of the JetStar was tested in the Calspan 2.44 m (8 ft) transonic wind tunnel in February 1981. Initially, both sides of the wing incorporated the same glove contours, but only the left glove contained pressure taps. Provisions for mounting the McDonnell Douglas leading-edge shield were included in the right glove. Detailed model information and a description of the proposed test program is given in Reference 3. Various Lockheed personnel, John Allen from McDonnell Douglas, and Mike Fischer and Dr. Richard Whitcomb of NASA Langley monitored the test. A complete report of the test may be found in the final test report from Calspan, Reference 4. The main results are summarized in this report.

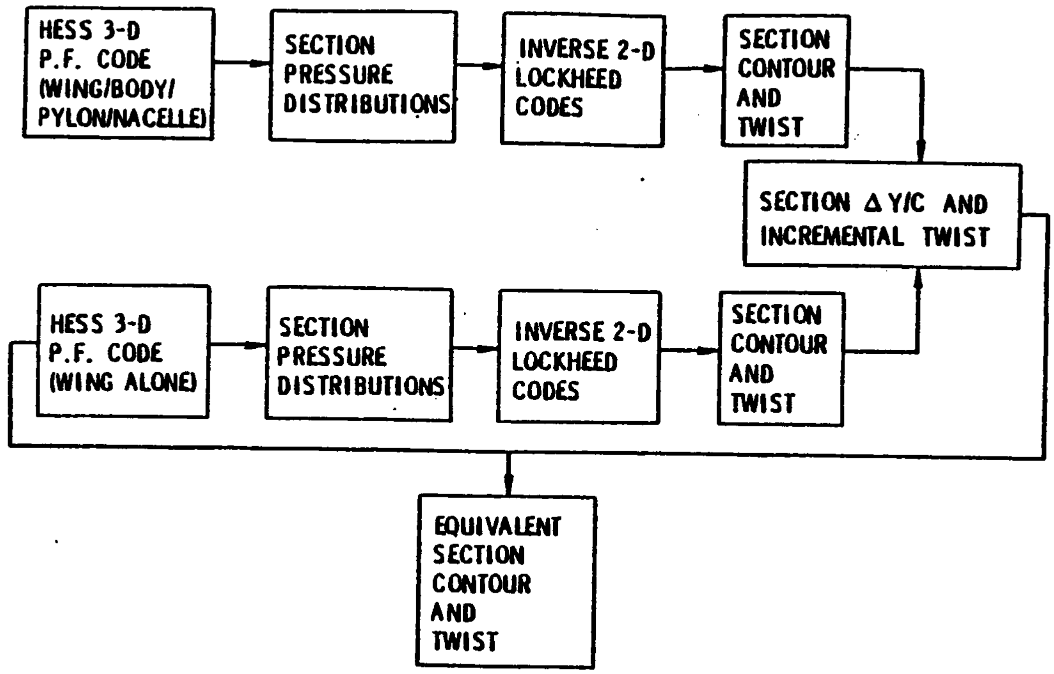

Figure 193. Equivalent Wing Method Schematic

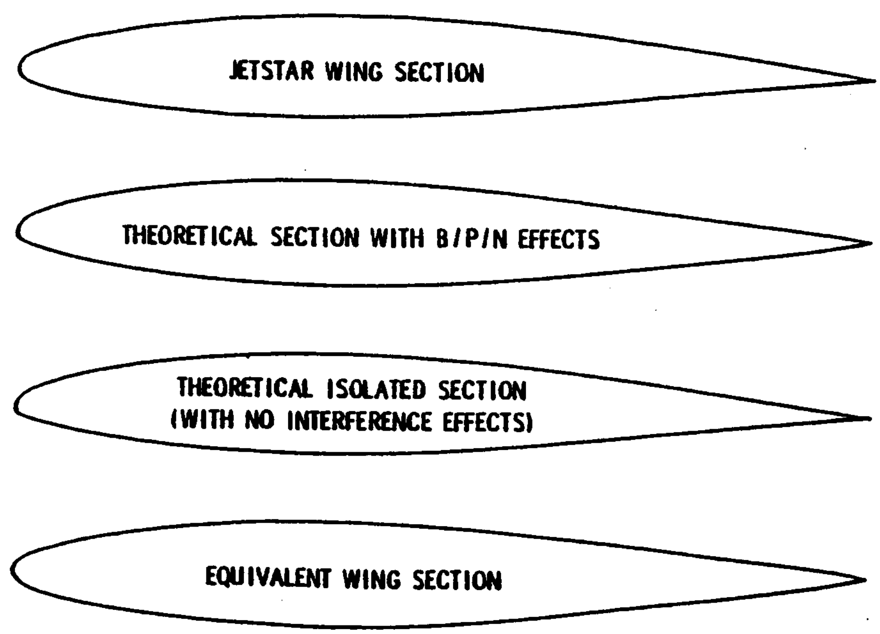

Figure 194. Equivalent Wing Section Comparison

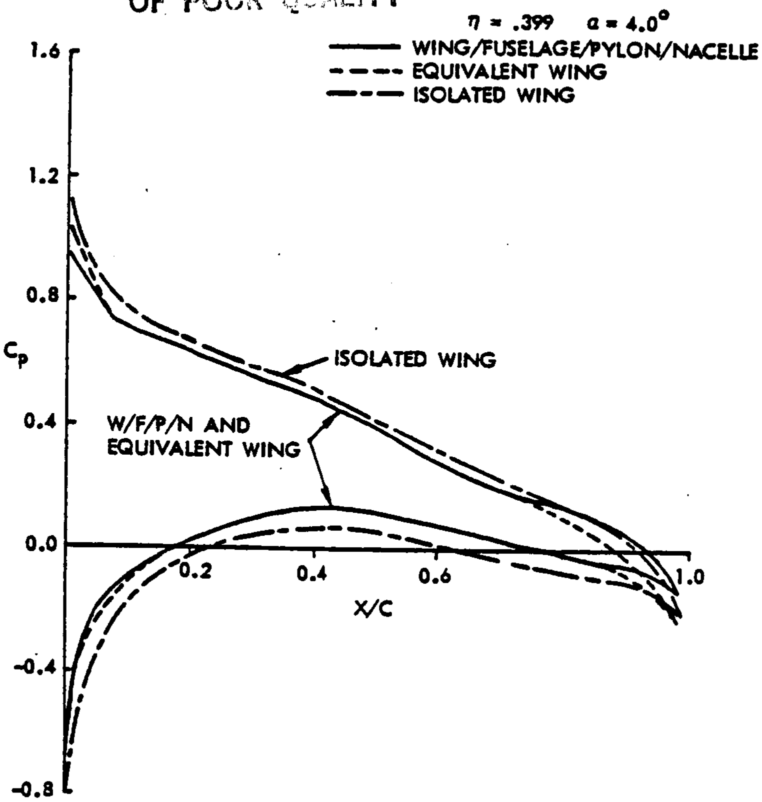

Figure 195. Hess Pressure Distribution Comparisons

7.3.1 CALSPAN Objectives

The objectives of the test were to:

(1) Obtain measured pressure distributions on the glove.

(2) Evaluate the effects of the McDonnell Douglas leading-edge shield.

(3) Evaluate the effects of removing the external fuel tanks on the aircraft handling qualities.

(4) Obtain tanks-off and leading-edge slats retracted force data.

The test run schedule is shown in Table 15. Most of the flaps down data were run at a Reynolds number of 6.56 x 10^6m (2 x 10^6/ ft) and 0.10 - 0.14 rad (6 - 8 deg) angle of attack.

7.3.2 Aircraft Stability and Control Results

The effects of the McDonnell Douglas leading-edge shield and an evaluation of the effects of removing the external fuel tanks are presented in a separate stability and control report, Reference 5. That report concludes that no significant stability and control problems are exhibited, and the aircraft should exhibit acceptable behavior throughout its flight regime. The following specific conclusions are taken from the report:

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TABLE 15. CALSPAN WIND TUNNEL TEST RUN SCHEDULE

CONFIGURATION									TEST CONDITIONS												REMARKS
										RUN NUMBER AT INDICATED MACH NUMBER											LFC JETSTAR
FUSELAGE	WING	PYLON	NACELLE	HORIZ. STAB.	VERT. STAB	FLAPS	GLOVE	SHIELD	R_N/FT x 10^{-6}	.20	.50	.60	.70	.72	.74	.75	.76	.77	.825	.85	
BASE DATA - FLAPS DOWN - TAIL OFF																					
B_1	W_9	K_{11}	N_{318}	—	—	F_{38}^{20}	X_{18}	—	—	5											CHECK OUT
B_1	W_9	K_{11}	N_{318}	—	—	F_{38}^{20}	X_{18}	—	2.0	5/6											
B_1	W_9	K_{11}	N_{318}	—	—	F_{38}^{50}	X_{18}	—	2.0	8/9											
SHIELD - FLAPS DOWN - TAIL OFF																					
B_1	W_9	K_{11}	N_{318}	—	—	F_{38}^{50}	X_{18}	Y_1^{90}	2.0	11/12											$\alpha_c = 18°$
B_1	W_9	K_{11}	N_{318}	—	—	F_{38}^{50}	X_{18}	Y_1^{122}	2.0	13											$\alpha_c = 18°$
B_1	W_9	K_{11}	N_{318}	—	—	F_{38}^{20}	X_{18}	Y_1^{122}	2.0	14/15											$\alpha_c = 18°$
B_1	W_9	K_{11}	N_{318}	—	—	F_{38}^{20}	X_{18}	Y_1^{122}	2.0	16											GRIT REMOVED FROM WING LOWER SURFACE
SHIELD - FLAPS DOWN - TAIL ON																					
B_1	W_9	K_{11}	N_{318}	H_1^0	V_1	F_{38}^{20}	X_{18}	Y_1^{122}	2.0	18											
B_1	W_9	K_{11}	N_{318}	H_1^1	V_1	F_{38}^{20}	X_{18}	Y_1^{122}	2.0	19											
B_1	W_9	K_{11}	N_{318}	H_1^{-9}	V_1	F_{38}^{20}	X_{18}	Y_1^{122}	2.0	20											
B_1	W_9	K_{11}	N_{318}	H_1^{-9}	V_1	F_{38}^{50}	X_{18}	Y_1^{122}	2.0	22/23											
B_1	W_9	K_{11}	N_{318}	H_1^0	V_1	F_{38}^{50}	X_{18}	Y_1^{122}	2.0	24/25											
TRIM DATA - 50° FLAPS																					
B_1	W_9	K_{11}	N_{318}	H_1^0	V_1	F_{38}^{50}	X_{18}	—	2.0	26											
TRIM DATA - 20° FLAPS																					
B_1	W_9	K_{11}	N_{318}	H_1^0	V_1	F_{38}^{20}	X_{18}	—	2.0	28/29											
B_1	W_9	K_{11}	N_{318}	$H_1^{-4.25}$	V_1	F_{38}^{20}	X_{18}	Y_1^{122}	2.0	30											
B_1	W_9	K_{11}	N_{318}	$H_1^{-4.25}$	V_1	F_{38}^{20}	X_{18}	Y_1^{122}	3.0	31											
B_1	W_9	K_{11}	N_{318}	$H_1^{-4.25}$	V_1	F_{38}^{20}	X_{18}	Y_2^{122}	3.0	32											CUT-OFF SHIELD
B_1	W_9	K_{11}	N_{318}	$H_1^{-4.25}$	V_1	F_{38}^{20}	X_{18}	Y_2^{122}	3.0	33											FIXED TRANSITION
SHIELD - FLAPS UP - TAIL ON																					
B_1	W_9	K_{11}	N_{318}	H_1^0	V_1	—	X_{18}	Y_2^{122}	3.0	34	38		37		36	35					NEW FLAP
B_1	W_9	K_{11}	N_{318}	H_1^0	V_1	—	X_{18}	—	3.0	39	43		42		41	40					NEW FLAP
TRIM DATA - FLAPS UP																					
B_1	W_9	K_{11}	N_{318}	$H_1^{-4.25}$	V_1	—	X_{18}	—	3.0	52			49/50			48	47	46	45	44	OLD FLAP
B_1	W_9	K_{11}	N_{318}	$H_1^{-4.25}$	V_1	—	X_{18}	—	3.0	53			55			54					GRIT OFF LEFT GLOVE OLD FLAP
B_1	W_9	K_{11}	N_{318}	$H_1^{-4.25}$	V_1	—	X_{18}	—	3.0				61			60	59	58	57	56	OLD FLAP
NACELLE/PYLONS OFF - FLAPS UP - TAIL OFF																					
B_1	W_9	K_{11}	N_{318}	—	—	—	X_{18}	—	3.0	63			69		68	67	66	65	64		OLD FLAP
BASE DATA - FLAPS UP - TAIL OFF																					
B_1	W_9	—	—	—	—	—	X_{18}	—	3.0	71			77		76	75	74	73	72		
B_1	W_9	—	—	—	—	—	X_{18R} X_{19L}	—	3.0				83		82	81	80	79			RECONTOURED GLOVE LEFT SIDE

TABLE 15. CALSPAN WIND TUNNEL TEST RUN SCHEDULE (CONT'D)

CONFIGURATION										RUN NUMBER AT INDICATED MACH NUMBER												REMARKS
FUSELAGE	WING	PYLON	NACELLE	HORIZ. STAB.	VERT. STAB.	FLAPS	GLOVE	SHIELD	$R_N/FT \times 10^{-6}$	.20	.50	.60	.70	.72	.74	.75	.76	.77	.80	.825	.85	LFC JETSTAR
NACELLE/PYLONS ON - FLAPS UP - TAIL OFF																						
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{19L}	—	3.0	85			90		89	88	86	87				
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{20L}	—	3.0				95		96	92	93	94				RECONTOURED GLOVE LEFT SIDE X_{20L}
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{21L}	—	3.0			103	102		101	98	99	100				RECONTOURED GLOVE LEFT SIDE X_{21L}
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{22L}	—	3.0			109	107		108	104	105	106				RECONTOURED GLOVE LEFT SIDE X_{22L}
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{23L}	—	3.0				115		114	111	112	113				RECONTOURED GLOVE LEFT SIDE X_{23L}
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{24L}	—	3.0				120		119	116	117	118				RECONTOURED GLOVE LEFT SIDE X_{24L}
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{25L}	—	3.0				125		124	121	122	123				RECONTOURED GLOVE LEFT SIDE X_{25L}
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{26L}	—	3.0				120		119	116	117	118				RECONTOURED GLOVE LEFT SIDE X_{26L}
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{27L}	—	3.0				125		124	121	122	123				RECONTOURED GLOVE LEFT SIDE X_{27L}
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{28L}	—	3.0				130		129	126	127	128				RECONTOURED GLOVE LEFT SIDE X_{28L}
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{28L}	—	3.0				136		134	131/135	132	133				
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{28L}	—	3.0				138			137						
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{28L}	—	3.0							140						
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{28L}	—	3.0							145		144	143	142	141	REMOVE WING
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{28L}	—	3.0	156	155		154	153	152	151	150	149	148	147	146	REMOVE GRIT
B_1	W_9	—	—	—	—	—	X_{18R} X_{28L}	—	3.0	157			164		163	162	161	160				REMOVE PYLON/NACELLE

OIL FLOW

B_1	W_9	—	—	—	—	—	X_{18R} X_{28L}	—	3.0							165	166					OIL FLOW $\alpha = 4°$, $\alpha = 3°$
B_1	W_9	K_{11}	N_{31B}	—	—	—	X_{18R} X_{28L}	—	3.0							167	168					OIL FLOW $\alpha = 4°$

(1) Basic aerodynamic data indicates an increase in static longitudinal stability for the LFC configuration. This is most pronounced at low Mach numbers.

(2) Stabilizer and elevator effectiveness remains the same or shows a slight increase.

(3) Extension of the McDonnell Douglas leading-edge shield on just the right wing has only a small effect on aerodynamic characteristics. While some minor trim changes may be required with the shield extended, these should not present a problem to the pilot. No abrupt asymmetric stall will occur.

(4) Maximum lift coefficient is reduced slightly. The largest decrease occurs for the flaps-down configurations. High-lift performance is adequate to meet mission requirements.

(5) Aircraft stall characteristics are degraded due to locking the leading-edge devices in the retracted position. The LFC aircraft tends to generate a nose-up moment that becomes more pronounced as the center of gravity moves aft. Wind tunnel data indicates that the tail remains effective up to the highest angles of attack tested. Sufficient control power will be available for stall recovery.

(6) Smaller margins between activation of the stall warning device and actual aerodynamic stall exist for the LFC JetStar than for the base aircraft. Much of this decrease in stall warning is thought to be due to conservatism applied when deriving the full-scale lift curves.

(7) A center-of-gravity envelope of 20 to 39 percent mean aerodynamic chord (MAC) provides adequate controllability and positive stability for all anticipated flight conditions. Preliminary weight and balance estimates indicate that actual center-of-gravity locations will be in the range 24 to 28 percent MAC, well within acceptable limits.

(8) For a given weight and center-of-gravity location, the LFC aircraft requires more aircraft nose-up stabilizer trim, and flies at a slightly lower angle of attack than the base configuration.

(9) Failure of the emergency pitch trim system, resulting in severe mis-trim, requires light landing weights, aft center-of-gravities, increased landing speed, or a combination of the three.

(10) The LFC JetStar exhibits positive speed stability at high Mach number without use of the Mach trim compensator (MTC). Use of the MTC will help ensure that the aircraft is not over-speeded.

(11) Wind tunnel data indicates no unusual high-speed aerodynamic characteristics. Current flight limits are acceptable. The present setting for the overspeed warning at M = 0.82 or 180m/sec (350 KEAS) can be retained.

(12) Use of the autopilot should be acceptable with the LFC aircraft. Some adjustment of system gains may be desirable.

(13) Directional stability is increased for the modified aircraft. Air minimum control speed remains approximately constant.

(14) No major changes in dynamic stability are evident. For the two modes where some noticeable changes may occur (i.e., rolling and dutch roll modes), the behavior of the LFC configuration will be improved.

(15) LFC stick and pedal forces will increase. For the applicable center-of-gravity range, the forces and gradients are acceptable.

7.3.3 Drag Polars

Figure 196 shows the full-scale lift curves trimmed at 25 percent MAC. Derivation of these curves is found in Reference 5. Trimmed full-scale drag polars are shown in Figure 197. These polars are based on test data adjusted for the same trim and full-scale lift increment as used for the lift curves. Figures 196 and 197 formed the basis for the take-off and landing data calculations, which are presented in Section 7.7.

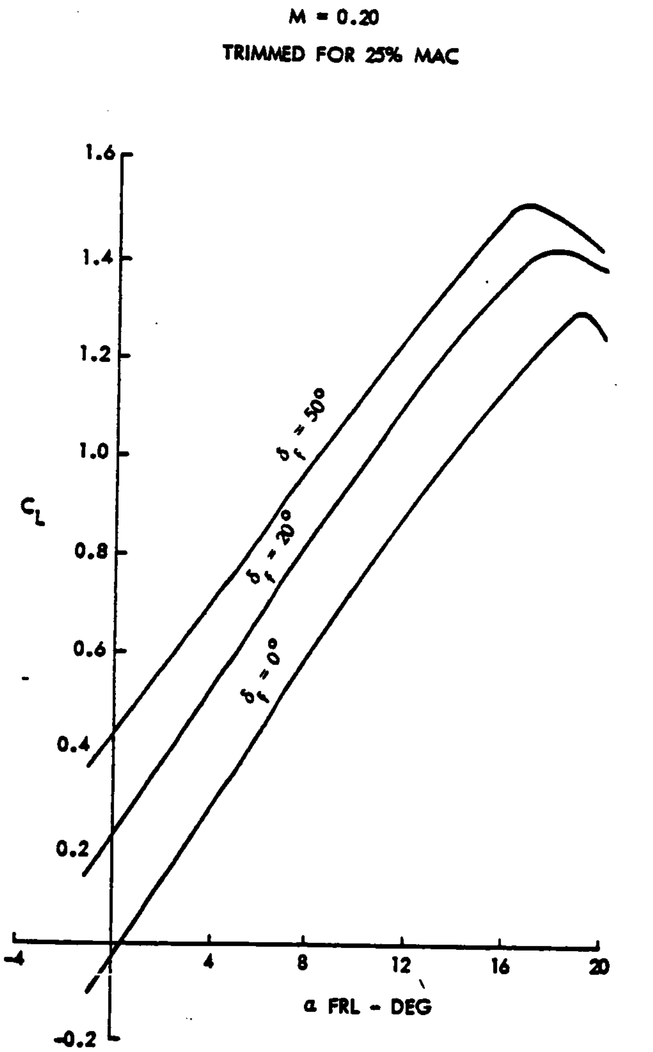

Figure 196. Trimmed Full Scale Lift Curves

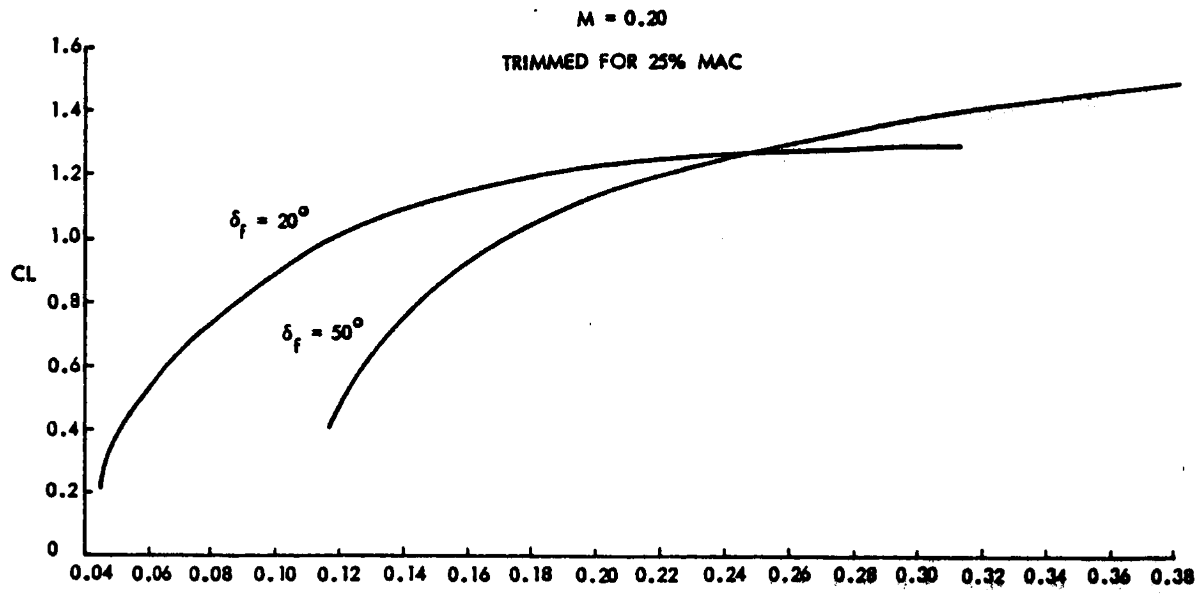

Figure 197. Trimmed Full Scale Drag Polars

7.3.4 Pressure Distributions

A major goal of the wind-tunnel test was to obtain pressure distributions on the glove, which could then be correlated against theoretical data. This goal was eventually achieved. However, the initial glove contour had large deviations from the specified MOD7Q contour. The reasons for these deviations were never fully understood.

7.3.4.1 Glove Recontouring

Figure 198 shows the pressure distributions for the original contour. These pressures had a high forward spike and were very wavy. At this point it was decided to recontour the glove while in the tunnel. The glove was altered to try to match the MOD7Q templates. Some areas of the glove were filed while others were filled-in and smoothed. It proved difficult to position the templates accurately on the model while in the tunnel. Figure 199 shows the pressure distributions for this revised contour. The main effect was to smooth the pressure.

The data were scattered near the design goal that had once been set for MOD5. After a series of in-tunnel glove recontourings, coordinated by Dr. Whitcomb, an acceptable pressure distribution was achieved. Figure 200 shows these pressures for various stations across the span of the glove. The effects of the glove recontourings can be seen by comparing Figures 198 and 200.

7.3.4.2 Final Glove Contour - MOD8

When the test was completed, the wing was carefully measured on the Cordax machine at Lockheed. The final glove contour established during the test was given the designation MOD8. Subsequent minor modifications made analytically resulted in the final MOD8C glove contours. These modifications are detailed in the final glove studies section of this report.

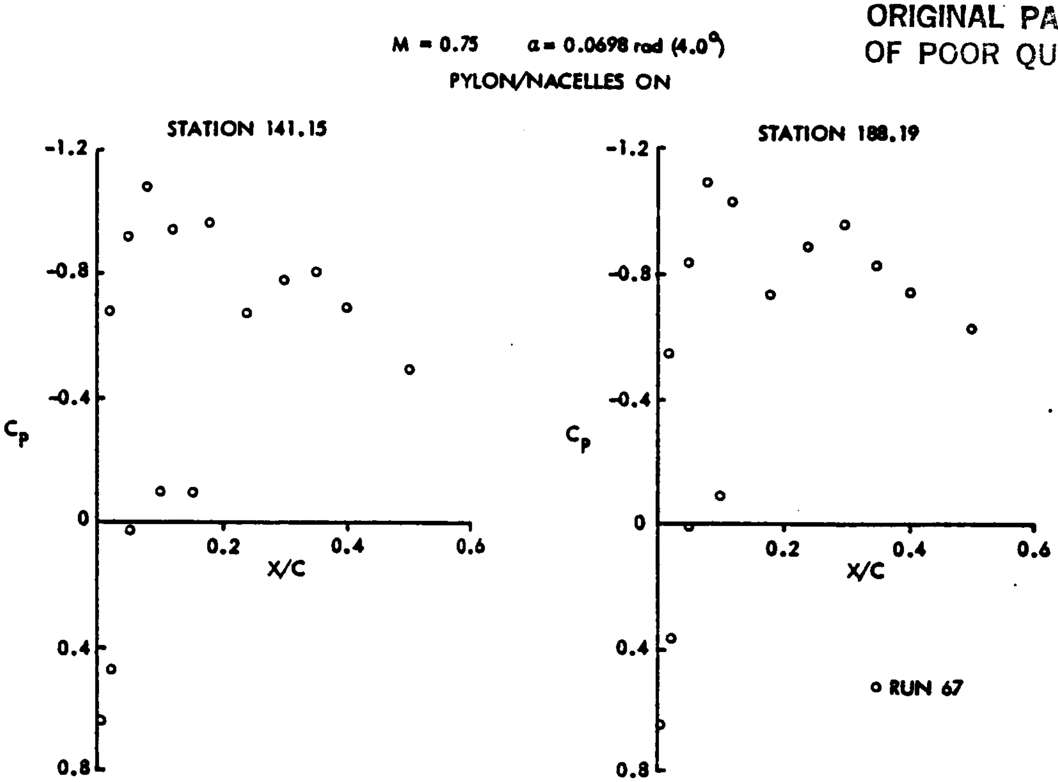

Figure 198. Original Contour Experimental Pressure Distributions

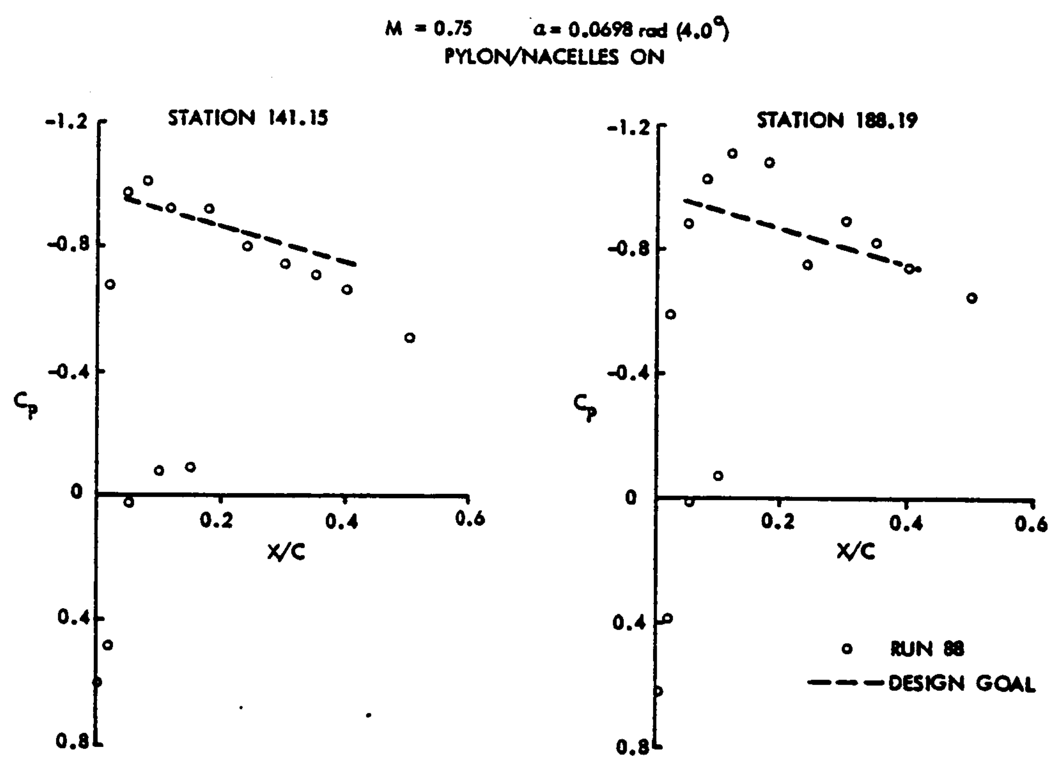

Figure 199. Interim Contour Experimental Pressure Distribution

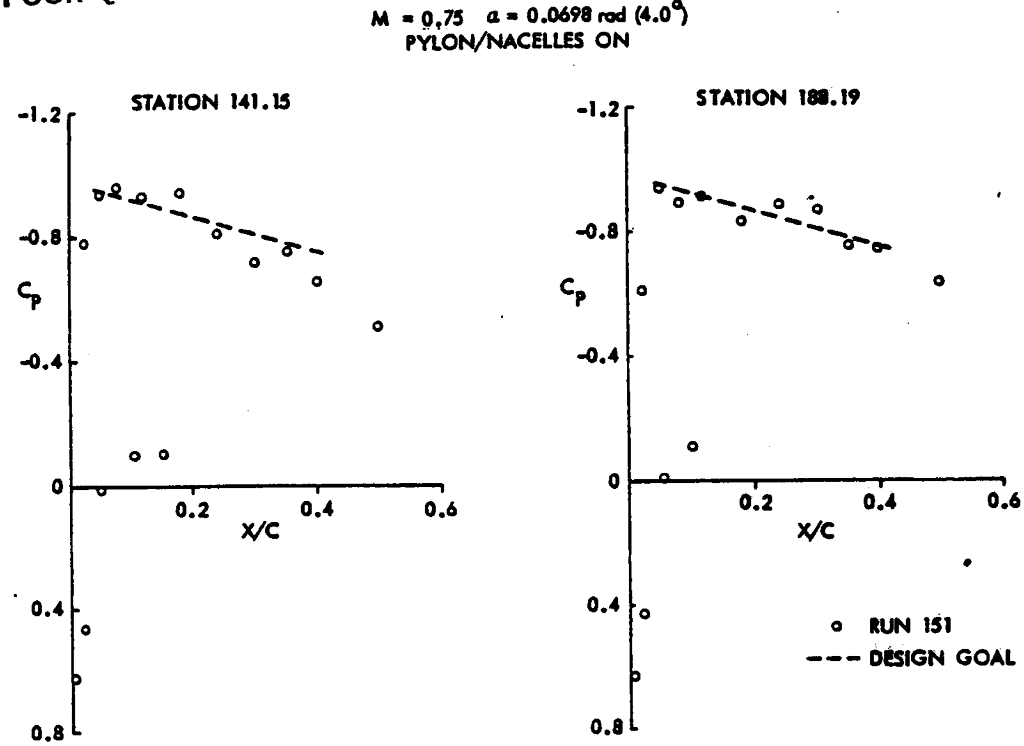

Figure 200. Final Contour (MOD8) Experimental Pressure Distributions

7.4 DATA CORRELATION

Theoretical pressure distributions for MOD8 were compared with experimental values to validate the equivalent wing methodology. Figures 201 and 202 compare equivalent wing theoretical pressures with experimental data for pylon/ nacelles off and on. The effect of the pylon/nacelles lowers the leading-edge pressure peak and moves the shock wave forward. The FL022NM equivalent wing pressures match quite well, especially back to 20 percent chord. This is aft of the region of active suction. The waviness of the FL022NM pressures is a result of the non-smooth contour produced by the in-tunnel recontouring. The code is very sensitive to non-smooth surfaces.

Figures 203 through 206 show the data correlation at various stations across the span of the glove. Generally, there is good agreement in the region of active suction. In the 30 percent chord region the equivalent wing pressures typically under-predict the correction for the pylon/nacelle effect. Toward the outboard portions of the glove this results in a favorable pressure gradient not demonstrated by the experimental data. In the next section of this report it will be shown that in spite of this theoretical natural-laminar-flow type of pressure distribution, the crossflow is large enough to cause transition near 5 percent chord. It must also be remembered that the experimental data did not have this natural-laminar-flow pressure distribution.

Good correlation of the equivalent wing pressures was also shown for off-design conditions. Figure 207 shows a comparison for an end-cruise

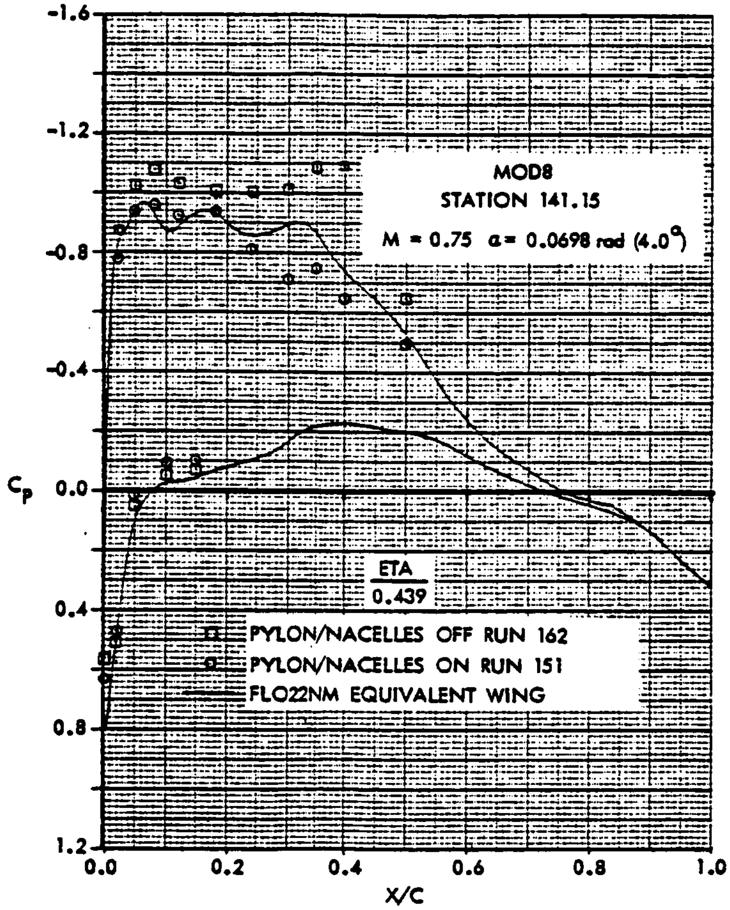

Figure 201. Effect of Pylons/Nacelles on Pressure Distribution, WS141.15

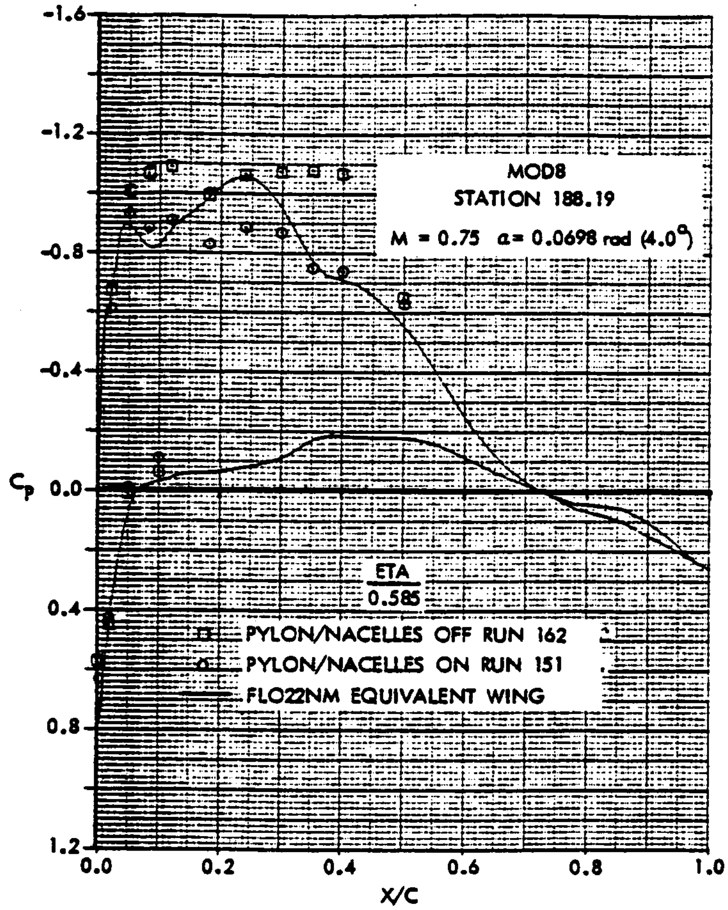

Figure 202. Effect of Pylons/Nacelles on Pressure Distribution, WS188.19

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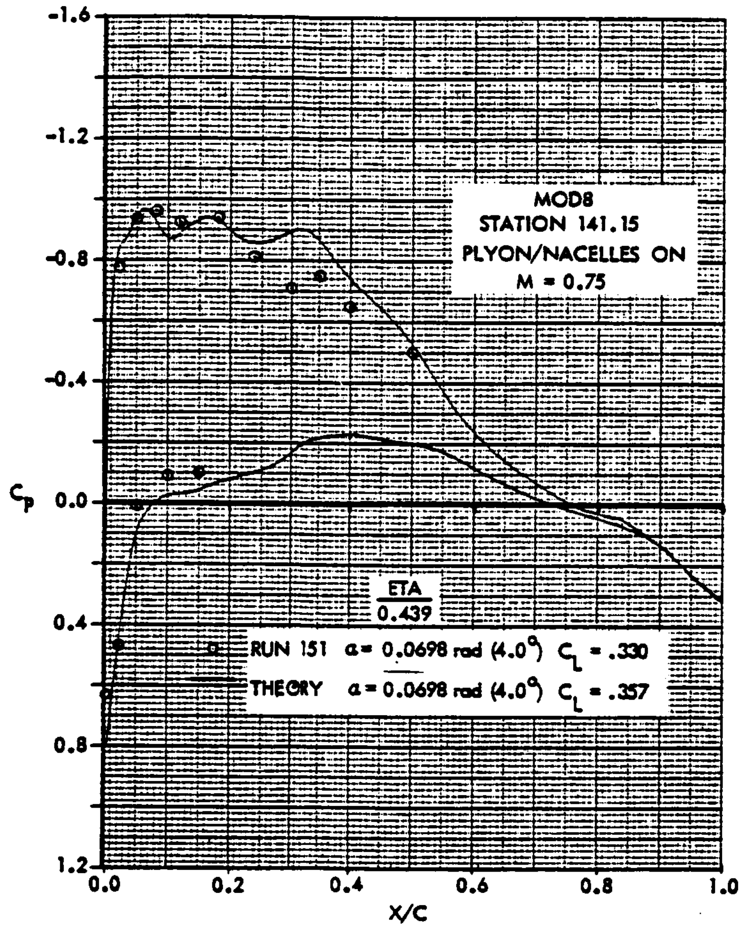

Figure 203. Correlation of Theoretical Pressures with Experimental Data, WS141.15

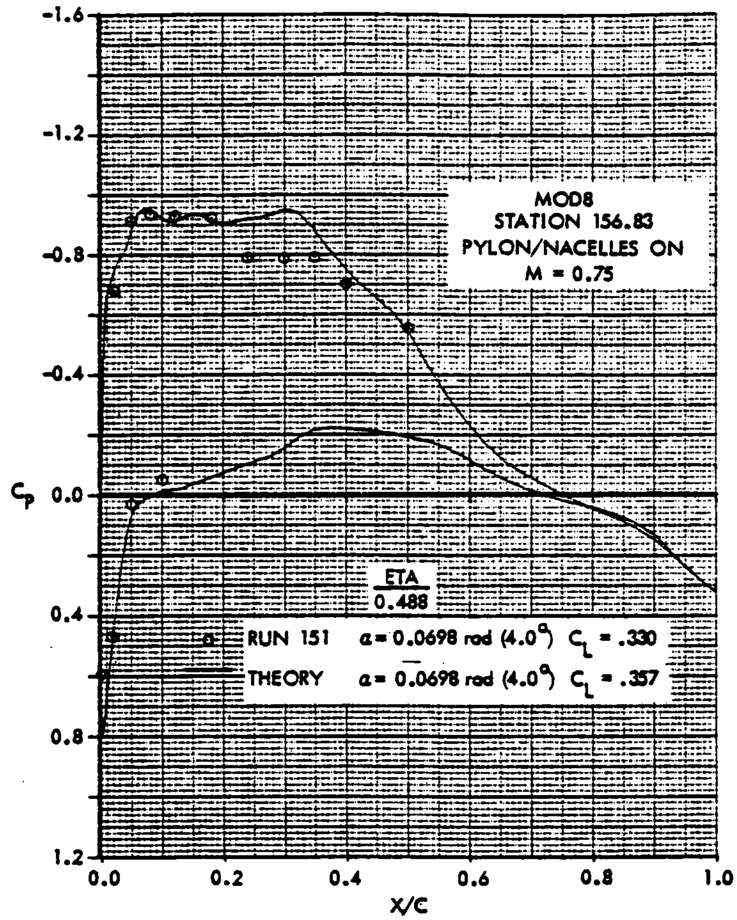

Figure 204. Correlation of Theoretical Pressures with Experimental Data, WS156.83

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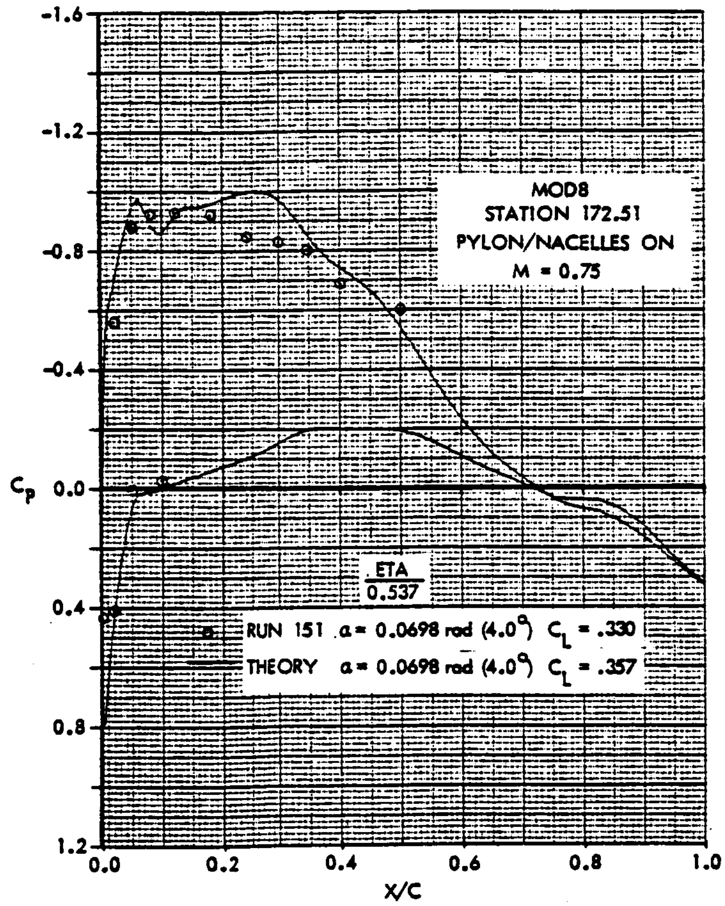

Figure 205. Correlation of Theoretical Pressures with Experimental Data, WS172.51

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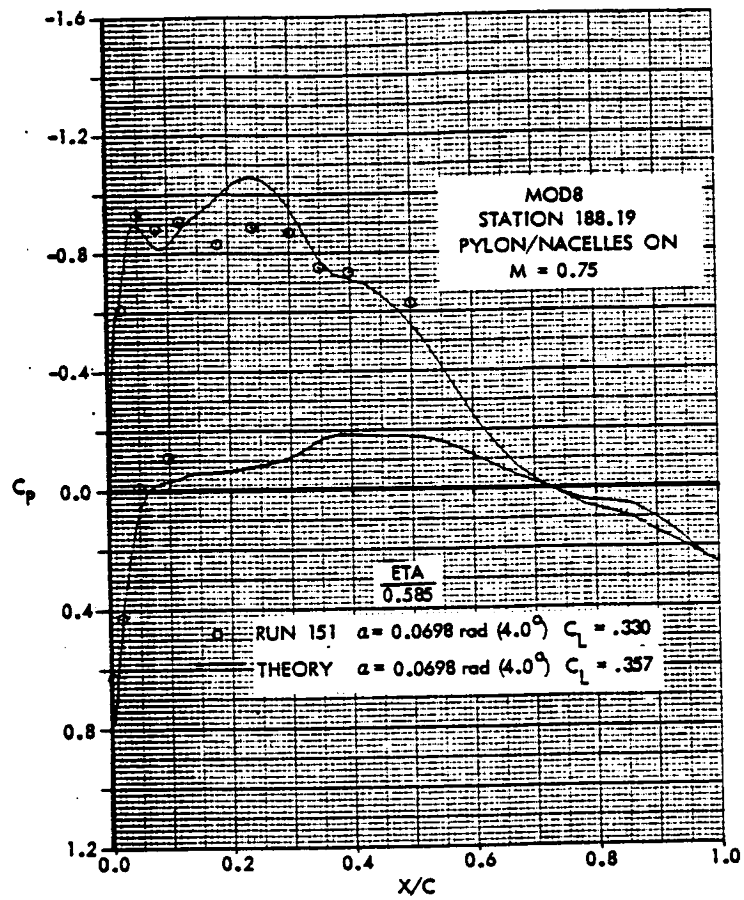

Figure 206. Correlation of Theoretical Pressures with Experimental Data, WS188.19

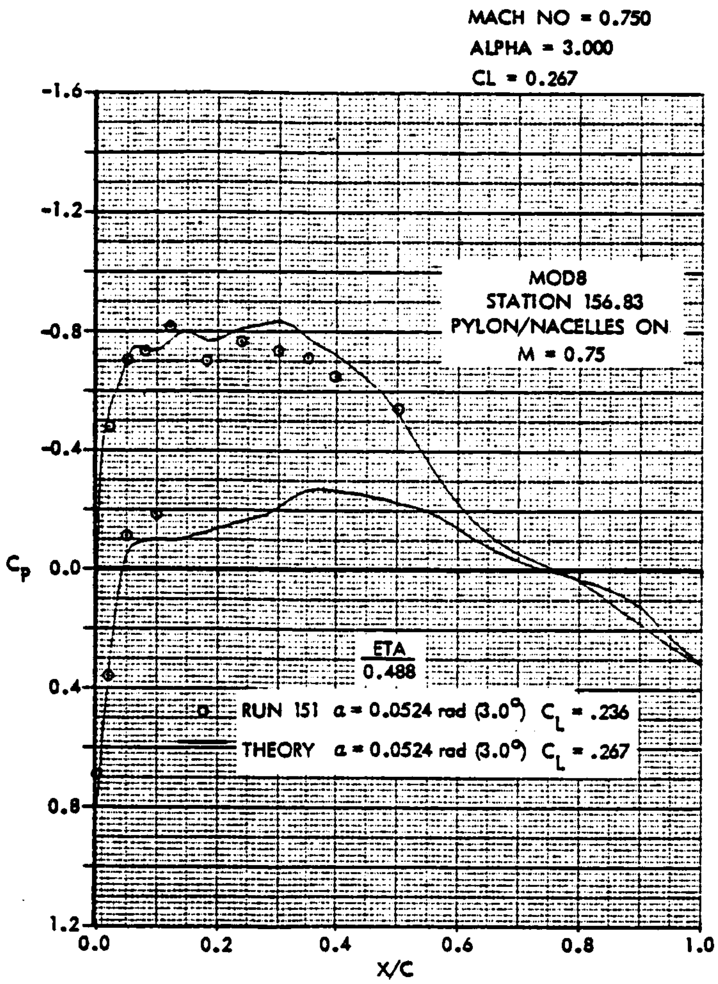

Figure 207. Correlation of Theoretical Pressures, End Cruise Condition

condition. Other comparisons are shown of Mach numbers 0.70 and 0.77 in Figures 208 and 209. In each case, good correlation is obtained for the extent of the active LFC region.

7.5 FINAL GLOVE (MOD8C) BOUNDARY LAYER STUDIES

Three minor changes were made to the measured wind tunnel glove contours to define the final MOD8C glove control stations, which are shown in Figure 210:

(1) The contours were smoothed analytically to reduce the waviness to the calculated pressure distributions.

(2) The outboard control station contour was altered slightly in the 2 percent chord region to decrease the spanwise pressure gradient, thus minimizing potential slot design problems.

(3) The glove portion of the contour was accurately fitted to the JetStar wing (as opposed to the model wing) by the loft group.

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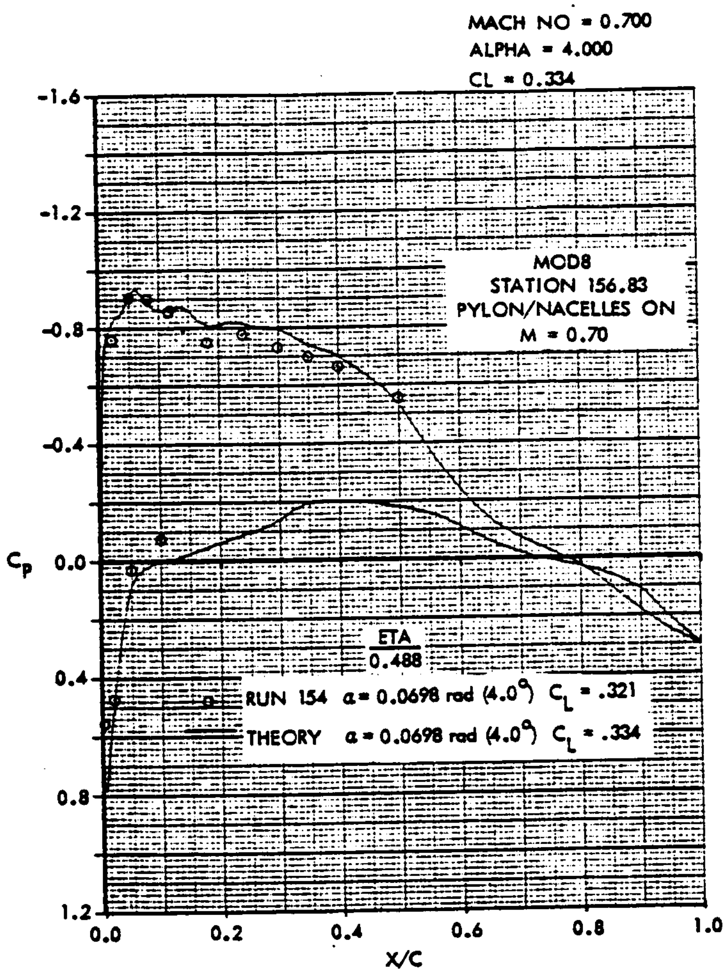

Figure 208. Correlation of Theoretical Pressures, M = 0.70

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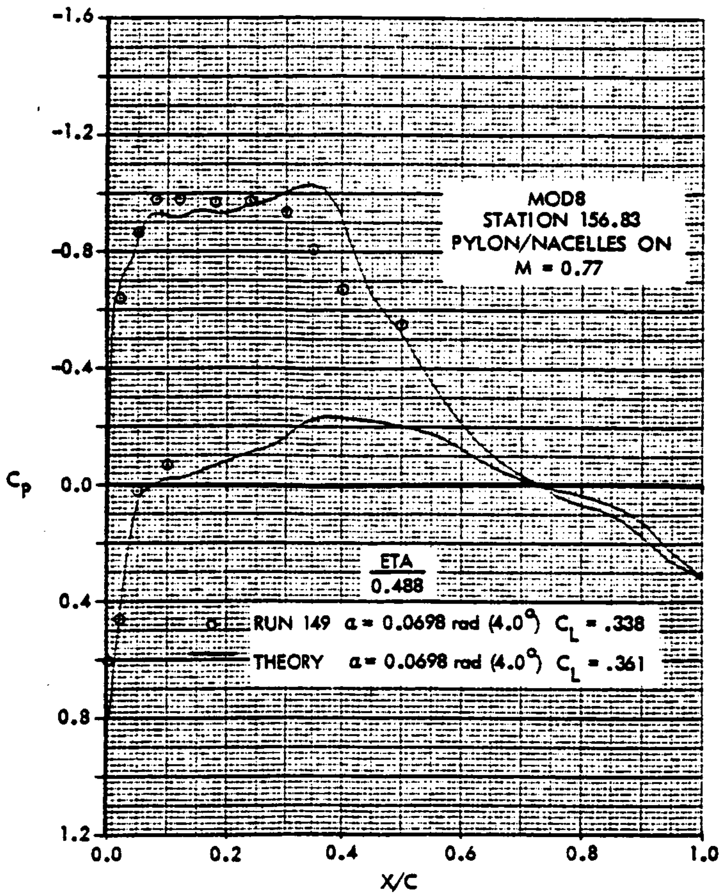

Figure 209. Correlation of Theoretical Pressures, M = 0.77

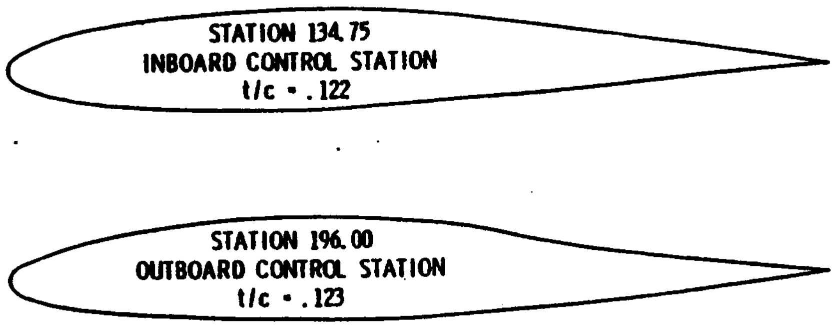

Figure 210. Final Glove (MOD8C) Control Stations

Design point pressure distributions for the MOD8C glove, as calculated by the FL022NM code, are shown in Figure 211. Flight conditions are: M = 0.75, altitude = 11,582m (38,000 ft), aircraft weight = 13,159 kg (29,000 lb).

Required suction distributions were established for these conditions and pressures.

7.5.1 Stagnation Point Calculation

The CEBECI code (Reference 6), which calculates laminar boundary layer data, and the SALLY code (Reference 7), which calculates the boundary layer stability data, were used in the boundary layer analyses. These codes were

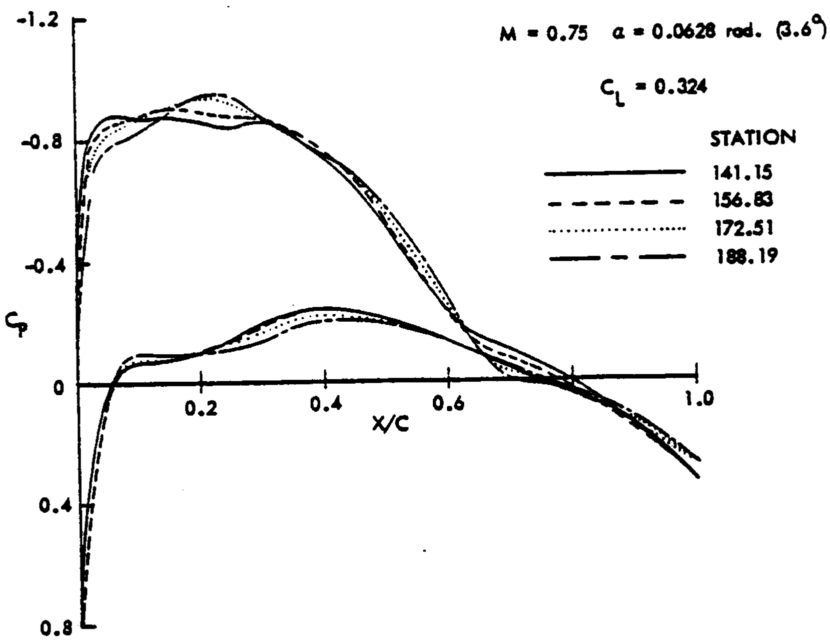

Figure 211. Mid-Cruise Pressure Distributions

found to be very sensitive to the values selected for the stagnation point. The method used to determine the stagnation data is shown in Figure 212. The pressures from FL022NM are plotted against $\sqrt{x/c}$ to expand the scale, and a smooth curve is faired through the data. The zero slope tangent to the point of highest pressure defines the stagnation values and location.

7.5.2 No-Suction Stability

Stability calculation results for a no-suction case, using the envelope method in the SALLY code, are shown in Figure 213. If a critical N factor value of 11 is assumed, transition is indicated near 5 percent chord on the upper surface and 4 percent on the lower. This analysis assumes a perfectly smooth surface with no slots. However, in flight with no suction applied, the presence of the slots in the leading-edge test article would probably "trip" the boundary layer forward of the predicted locations. For an operational LFC wing, it would be desirable to delay the need for suction as far aft as feasible. However, to prove the concept of LFC in this glove, the forward location of transition is necessary.

7.5.3 Suction Distribution

Initial attempts at defining the required suction distribution were based on delaying the start of suction past 1 percent chord. This was desired because of the limited volume in the JetStar wing leading edge. A suction distribution

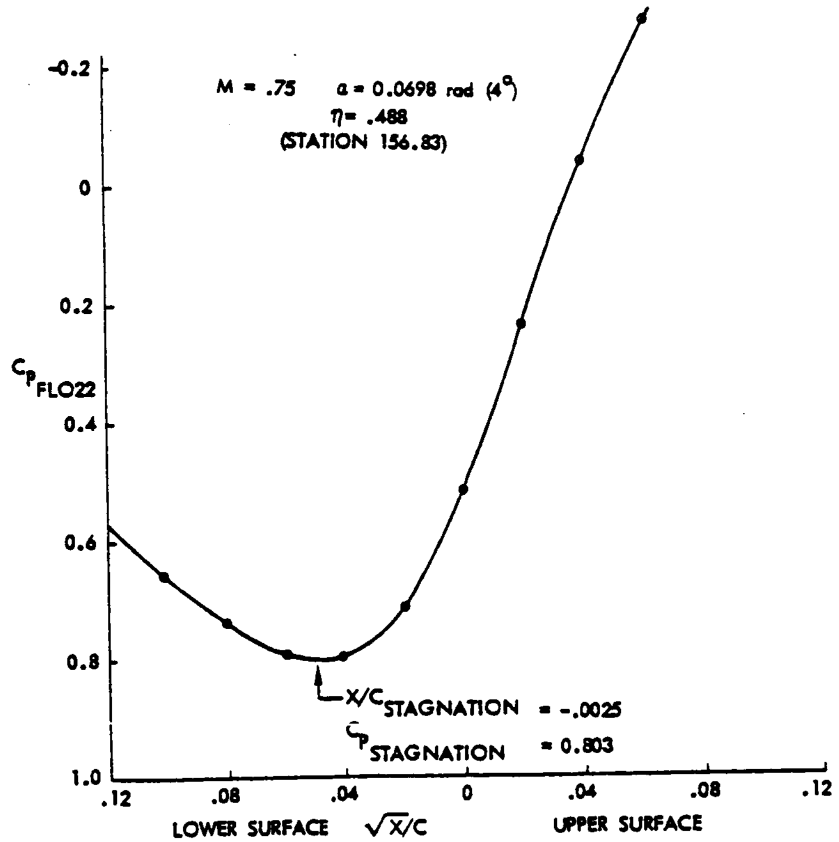

Figure 212. Stagnation Location Method Example

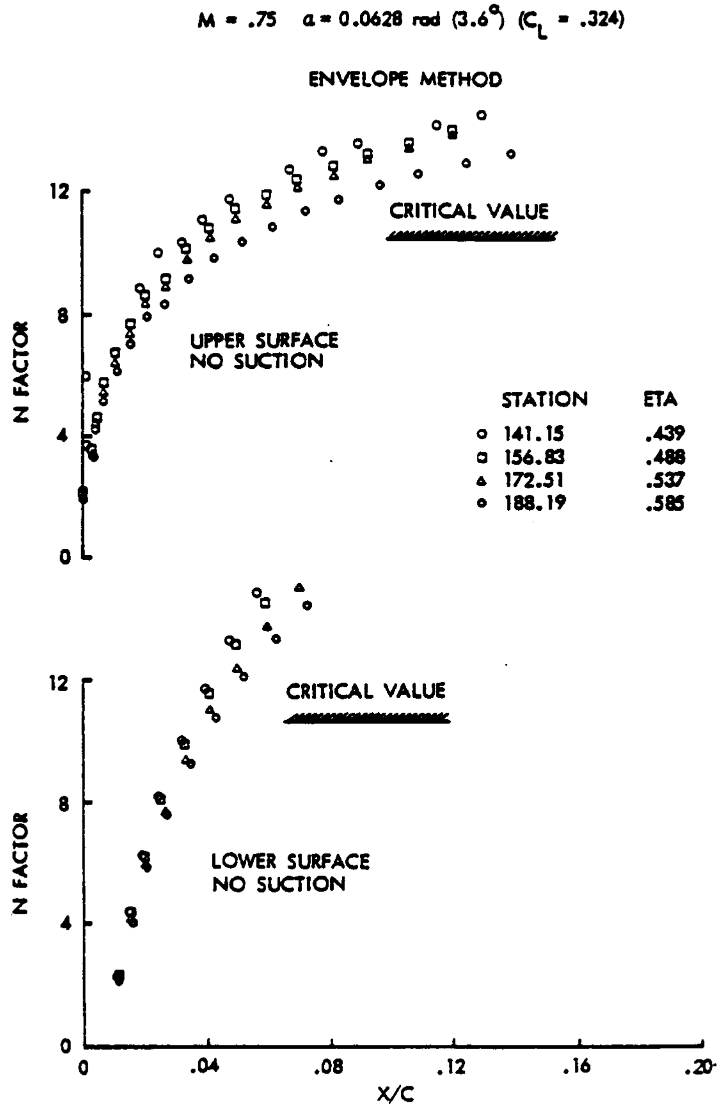

Figure 213. No Suction Crossflow N Factors, Envelope Method

meeting this requirement was established, but the distribution had high suction levels and a lengthy chordwise extent. These characteristics caused the capability of available suction pumps to become a limiting factor. The suction distribution shown in Figure 214 was established to alleviate this problem. Suction started at the nose of the section, and peaked at C_q = -0.0006 only in the forward few percent. Aft of that, the suction level drops to C_q = 0.0001 for the rest of the active section.

7.5.4 Crossflow N Factors

Crossflow N factors for the mid-cruise design point are shown in Figure 215 as calculated by the envelope method, and in Figure 216 as calculated by the fixed wavelength method. Several of these type plots are presented. Each figure shows the assumed critical N factor value: 11 for the envelope method and 8 for the fixed wavelength method. When appropriate, each figure also shows

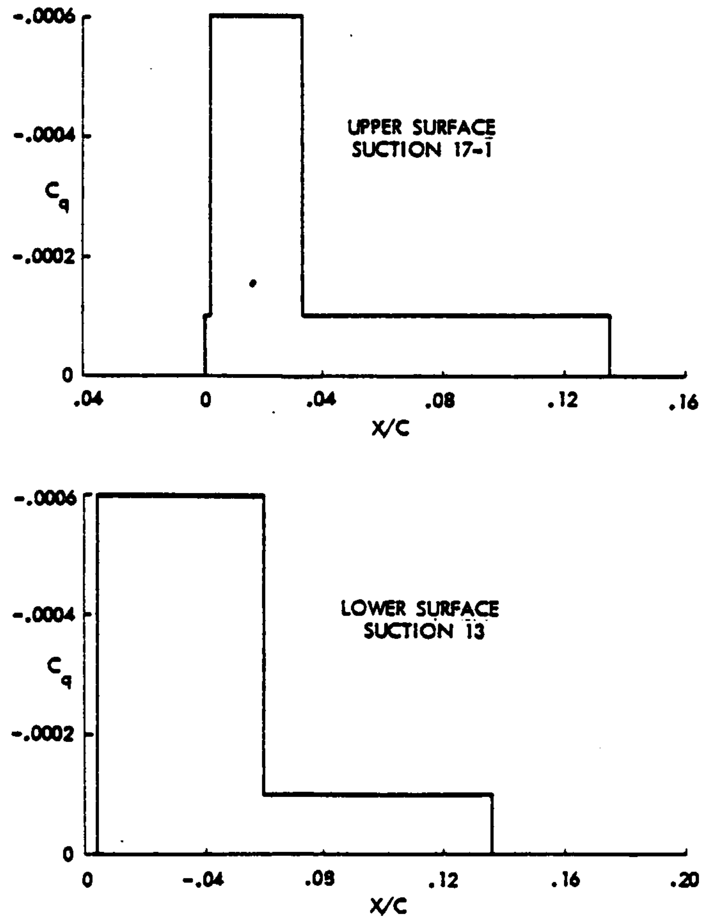

Figure 214. Required Suction Distributions

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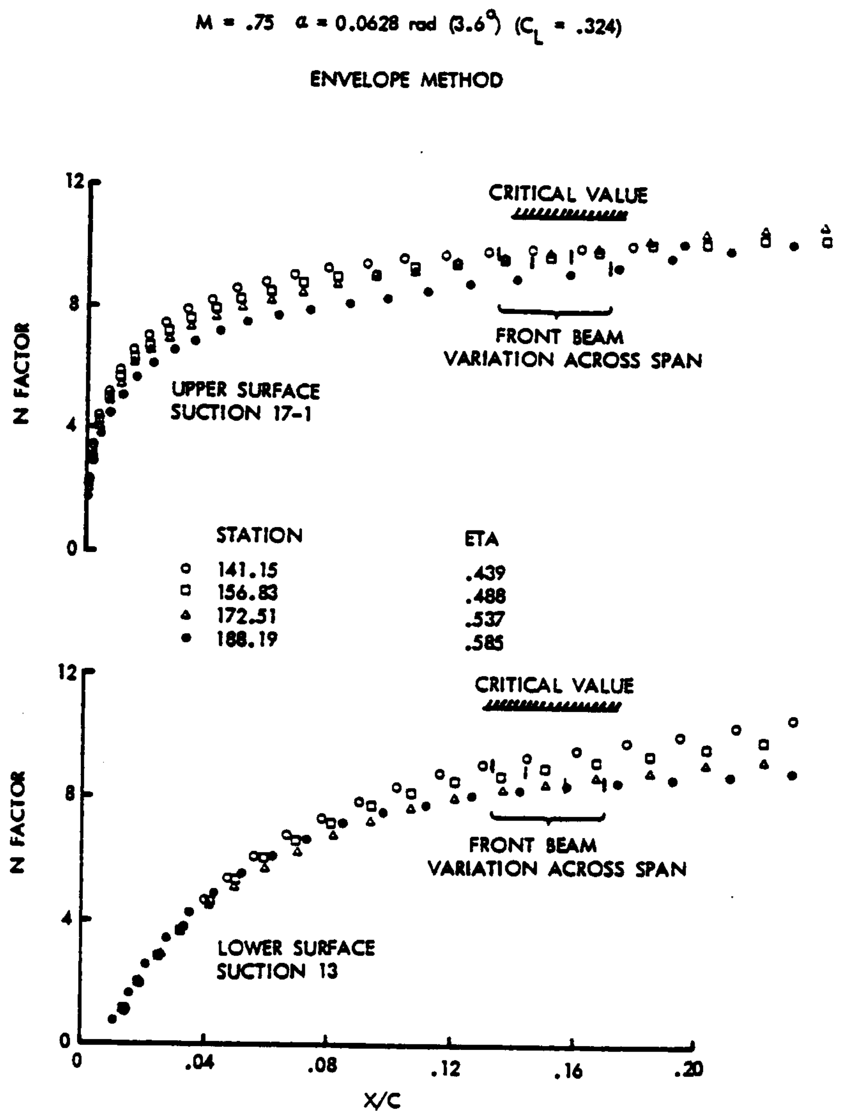

Figure 215. Mid-Cruise Crossflow N Factors, Envelope Method

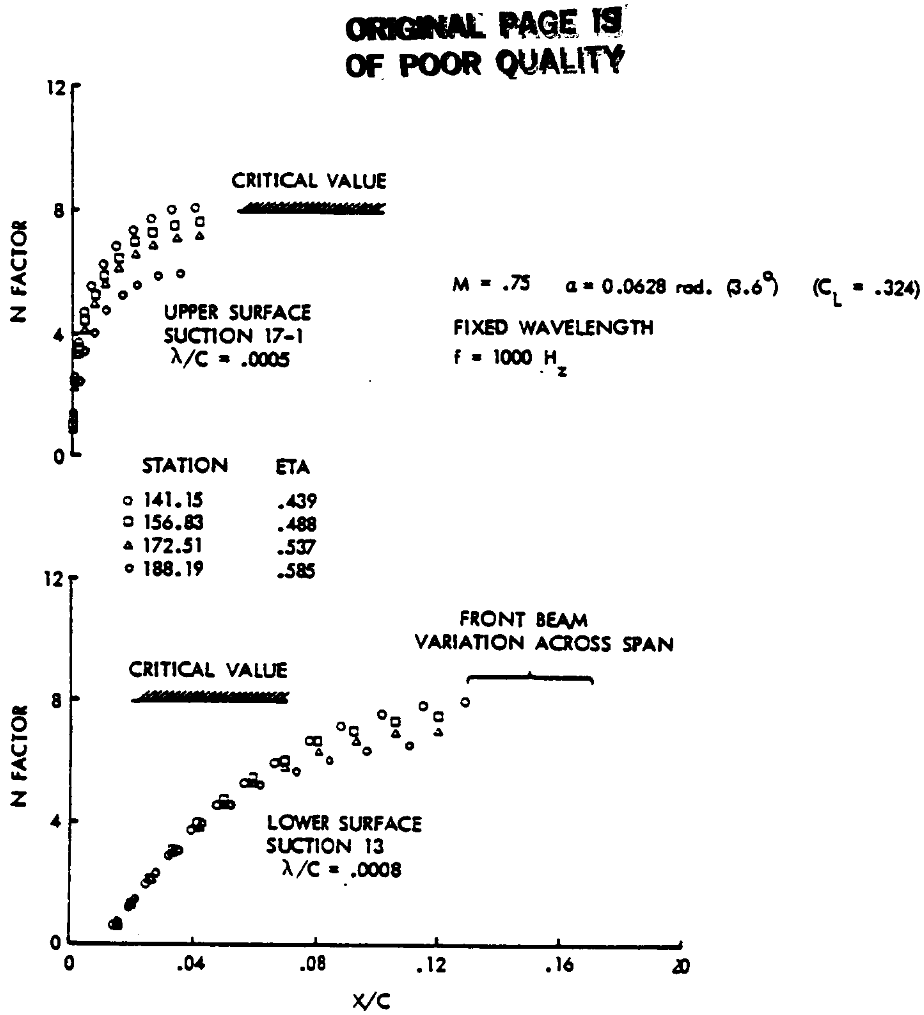

Figure 216. Mid-Cruise Crossflow N Factors, Fixed Wavelength Method

the front beam location across the span of the glove, which indicates the end of the suction region. For the fixed wavelength method, critical non-dimensionalized wavelengths were found to be 0.0005 and 0.0008 for the upper and lower surfaces, respectively.

The inboard station has the highest N factors for both glove surfaces. The suction was adjusted so that for the fixed wavelength method the critical value of 8 was just reached for the inboard station. This assures adequate suction for the outer portions of the glove. The N factors calculated by the envelope method never exceed a value of 10 in the active LFC region.

The suction system has an oversuction capability of 50 percent. For that level of oversuction, past Lockheed studies have shown an N factor decrease of 2.0 at 10 percent chord and 1.0 at 5 percent chord. This provides an adequate margin for the system.

7.5.5 Off-Design Suction

Having established a satisfactory suction distribution for the mid-cruise design point, off-design conditions at start and end-cruise, as well as other

Mach numbers and altitudes, were investigated. Figures 217 and 218 illustrate that the glove pressures behave well throughout the cruise regime. There is no appreciable change in the nature of the pressures from start-cruise to end-cruise.

The results of crossflow N factor calculations at start-cruise conditions are shown in Figures 219 and 220. Both calculation methods show the inboard station just reaching critical N factor values. The outboard station remains less critical. End-cruise results are shown in Figures 221 and 222. At this condition, the glove is substantially below critical N factor values.

It would be very useful to test the glove design at other than the M = 0.75. 11,582m (38,000 ft) design case. Calculations were made at the station just inboard of the glove center for flight conditions of M = 0.70 and 0.77 at 11,582 m (38,000 ft), and M = 0.75 at altitudes of 10,363 m (34,000 ft) and 9,144 m (30,000 ft). Figure 223 shows the pressure distributions for these conditions. Boundary layer stability results are shown in Figures 224 and 225. Both methods indicate that the established suction distribution is sufficient at 11,582 m (38,000 ft) to limit the N factor below critical values for the M = 0.70 to 0.77 range. However, while reducing altitude to 10,363 m (34,000 ft) seems to present no problems, critical N factors are exceeded at 9,144m (30,000 ft). This would be an interesting data point on which to gather information on the validity of the assumed critical N factor values and to test the over-suction capability of the glove.

Attachment line Reynolds numbers were calculated for the design and off-design conditions, and are listed in Table 16. At the design cruise condition, the values range from 89 to 112. Values at off-design conditions illustrate the Reynolds number increases possible by cruising at lower altitudes. The effects of cruising at alternate Mach numbers seem to be mixed and not obvious.

7.5.6 Tollmien-Schlichting Disturbances

The results of an investigation of Tollmien-Schlichting disturbances conducted at a station near the glove center are shown in Figure 226. A range of frequencies from 4,000 to 10,000 Hz was studied. In each case, if disturbances did start to amplify, it was aft of the front beam where there is no suction. This is consistent with previous Lockheed studies, which showed that moderate suction levels control Tollmien-Schlicting disturbances.

7.6 SUCTION ANALYSIS

7.6.1 Suction Distribution

The required suction distribution for the leading-edge test article was established from boundary layer stability considerations. For reference, stability calculations were made for a no-suction case using the "envelope" method. If a critical value of 11 is assumed, transition is indicated near 5 percent chord on the upper surface and 4 percent on the lower surface. For an operational wing, one would delay the need for suction as far aft as is feasible. However, to prove the concept of LFC on this glove, the forward location of transition is necessary.

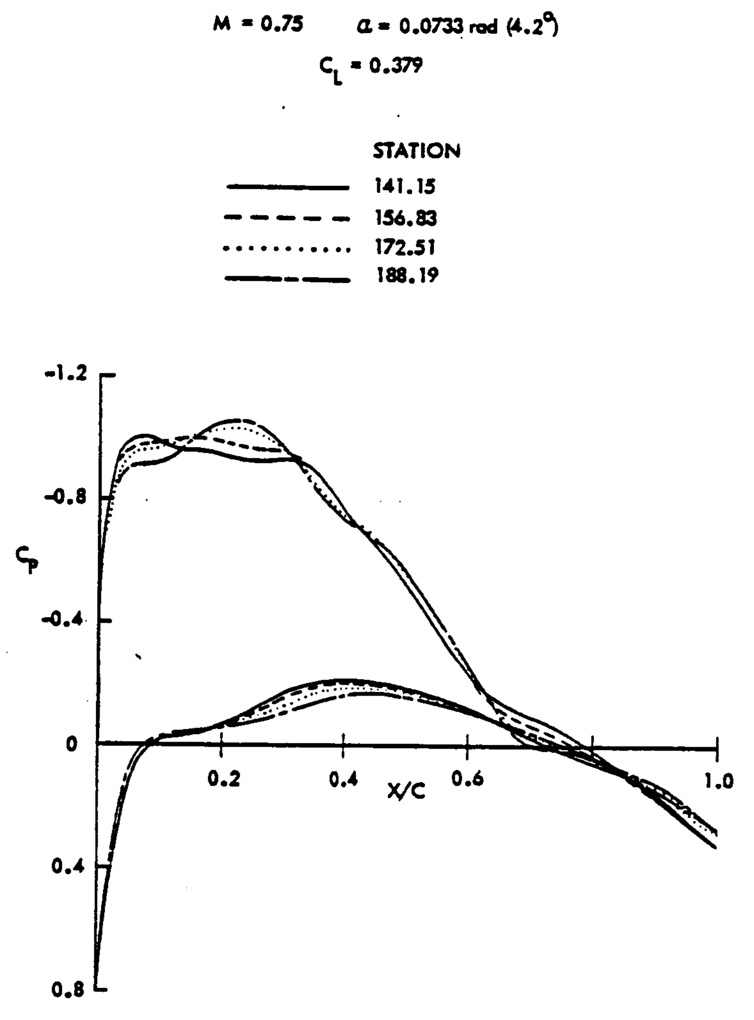

Figure 217. Start Cruise Pressure Distributions

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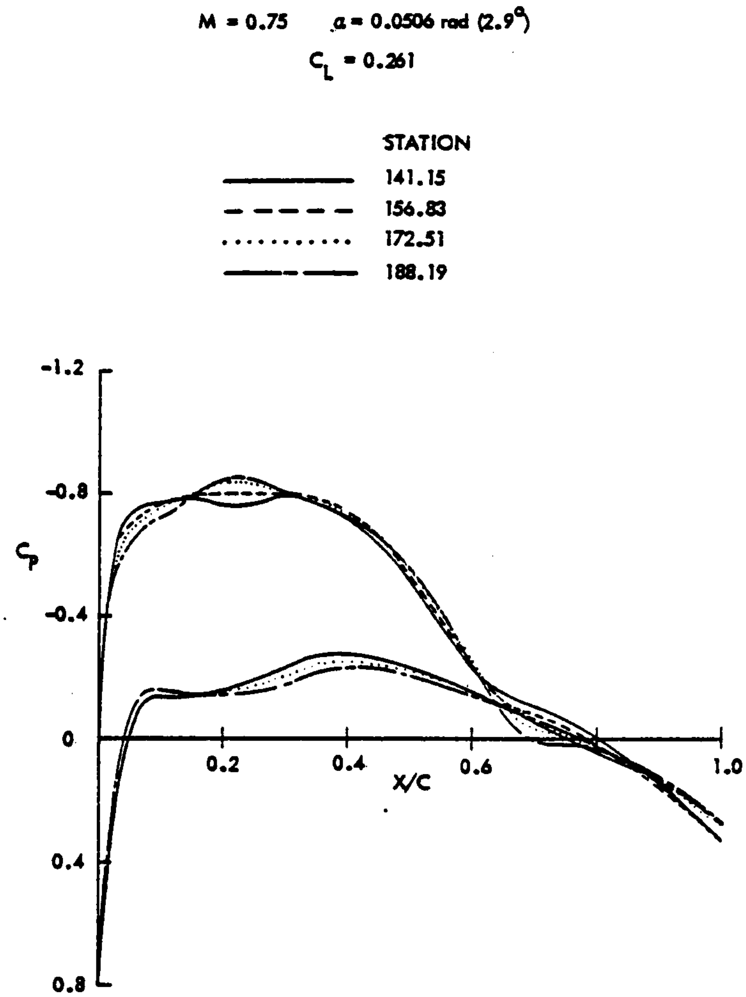

Figure 218. End-Cruise Pressure Distributions

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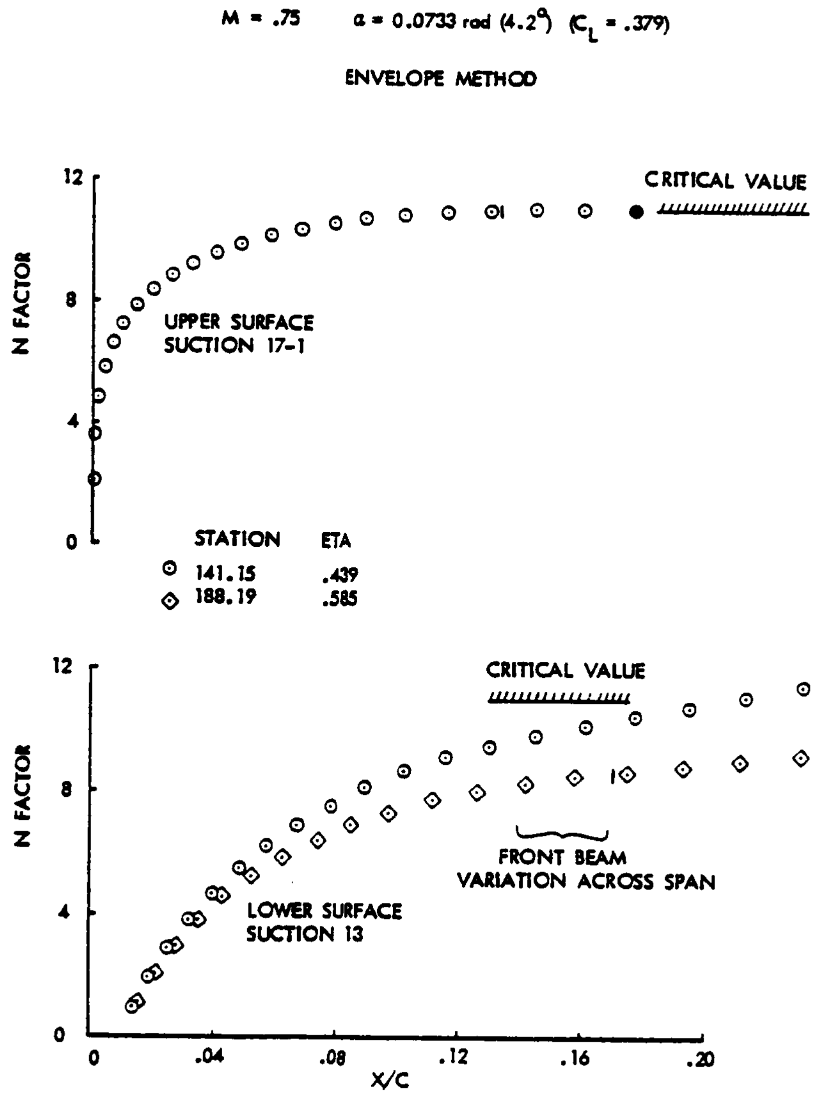


Figure 219. Start-Cruise Crossflow N Factors, Envelope Method

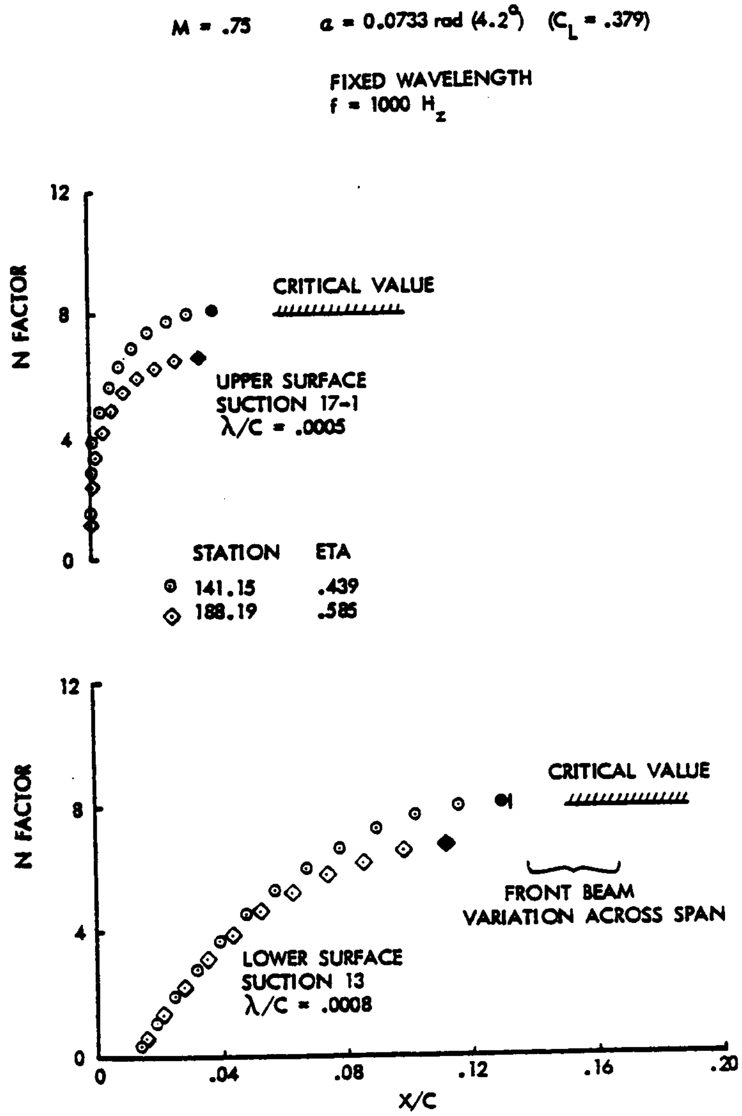


Figure 220. Start-Cruise Crossflow N Factors, Fixed Wavelength Method

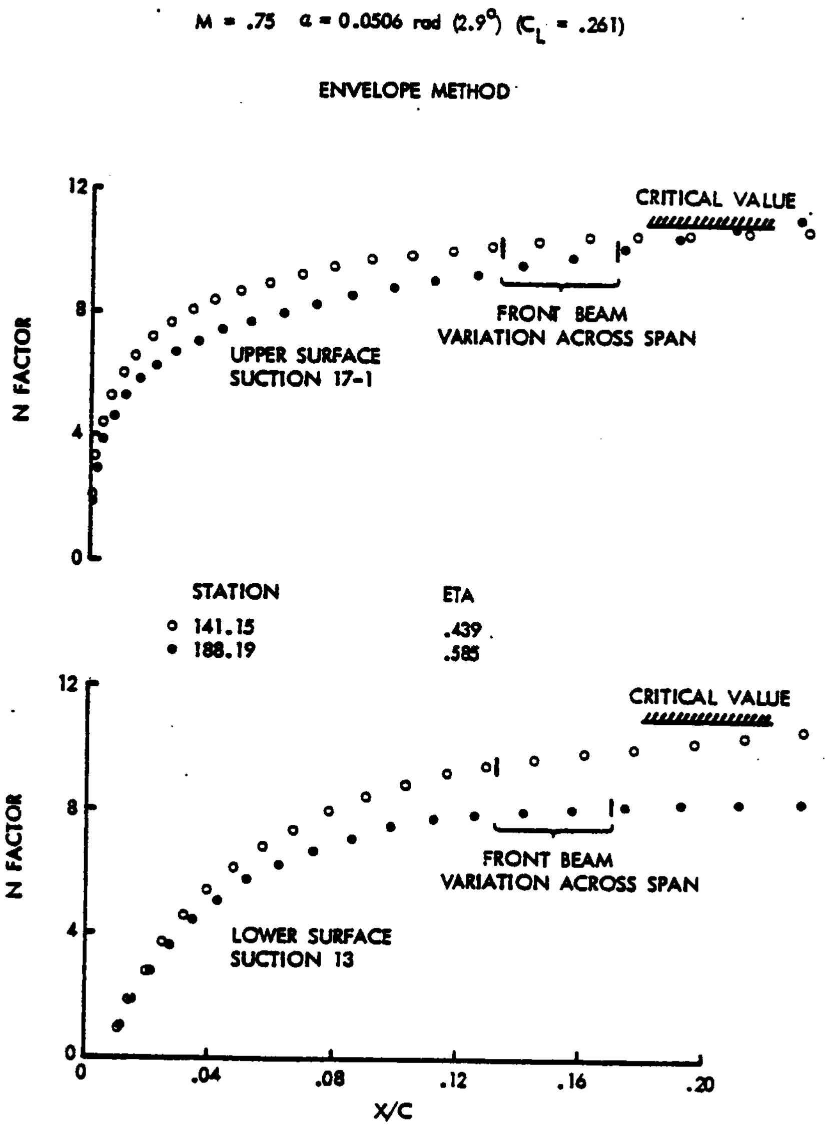


Figure 221. End-Cruise Crossflow N Factors, Envelope Method

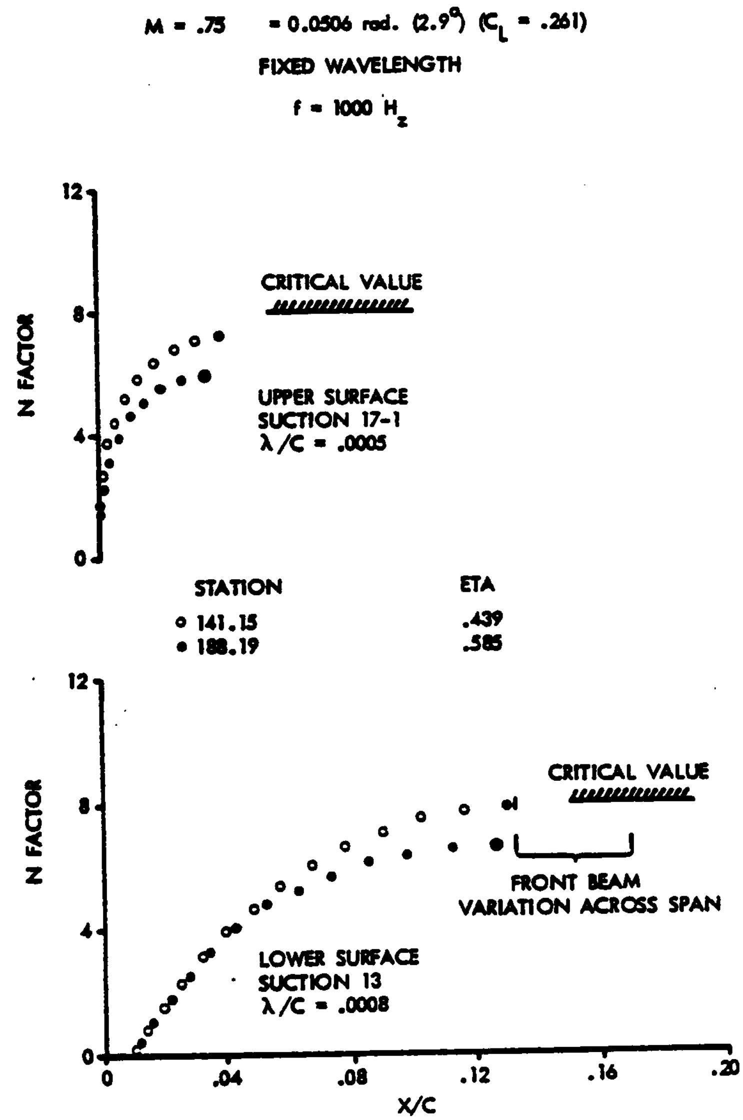


Figure 222. End-Cruise Crossflow N Factors, Fixed Wavelength

WING STATION 156.83
WT = 13,154 KG (29,000 LB.)

	M	α	ALT	rad	M
————	.77	3.3	38,000	0.0576	11,582
—— — ——	.70	4.2	38,000	0.0733	11,582
···········	.75	2.9	34,000	0.0506	10,363
– – – – –	.75	2.4	30,000	0.0419	9,144

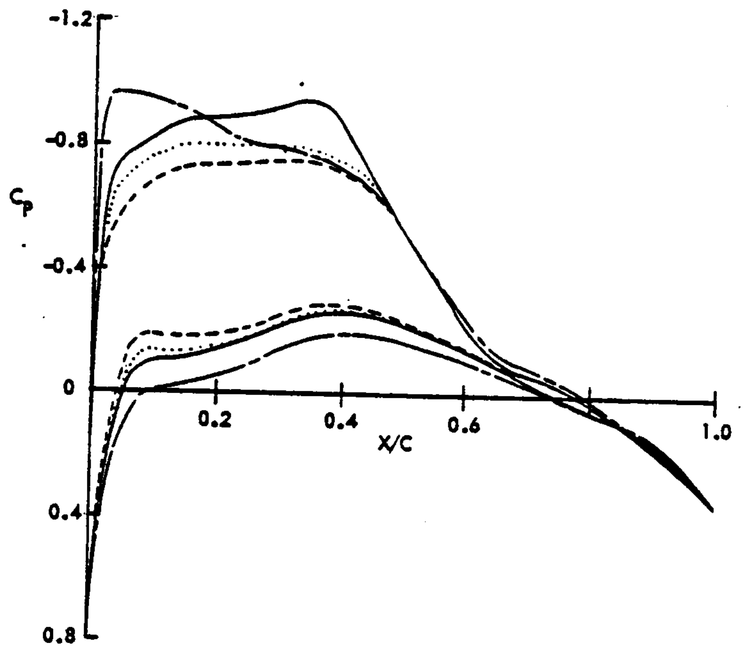


Figure 223. Off-Design Pressure Distributions

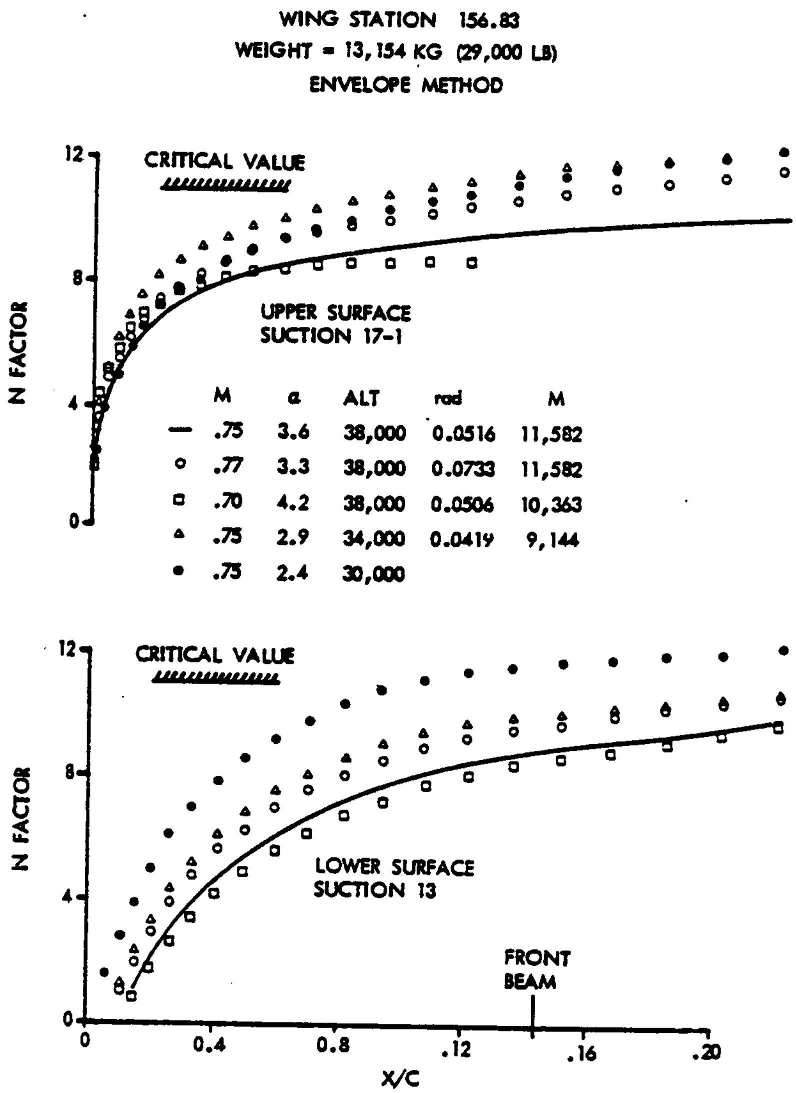
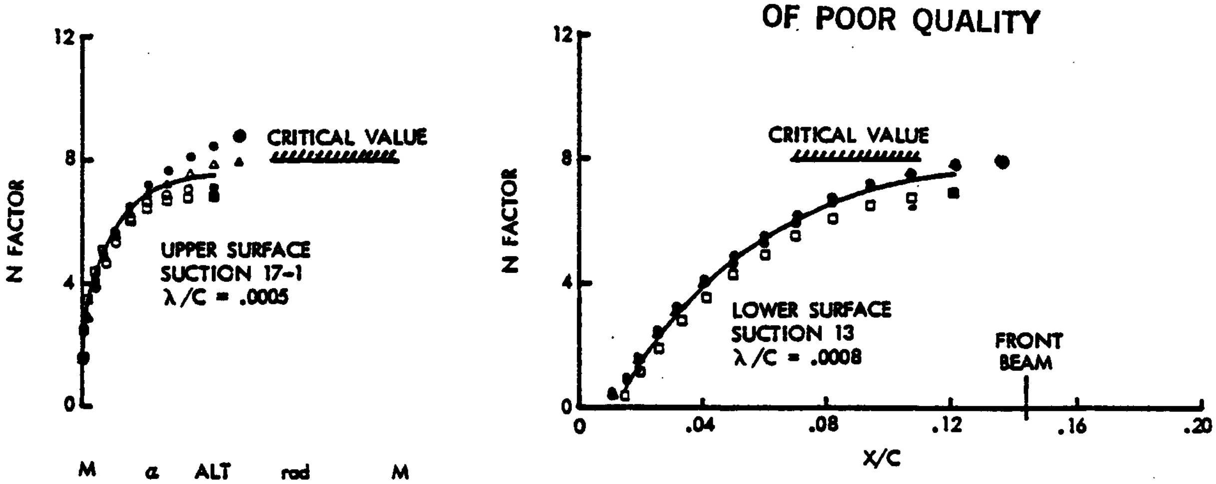


Figure 224. Off-Design Crossflow N Factors, Envelope Method



	M	α	ALT	rad	M
—	.75	3.6	38,000	0.0516	11,582
○	.77	3.3	38,000	0.0733	11,582
□	.70	4.2	38,000	0.0506	10,363
△	.75	2.9	34,000	0.0419	9,144
●	.75	2.4	30,000		

WING STATION 156.83
WEIGHT = 13,154 KG (29,000 LB)

FIXED WAVELENGTH
f = 1000 H_z

Figure 225. Off-Design Crossflow N Factors, Fixed Wavelength Method

TABLE 16. ATTACHMENT LINE REYNOLDS NUMBER

DESIGN POINT CONDITIONS

(M = 0.75, ALTITUDE = 11,582 M (38,000 FT)

	LOWER SURFACE				UPPER SURFACE			
W.S.	141.15	156.83	172.51	188.19	141.15	156.83	172.51	188.19
CRUISE CONDITION								
START	111.9			100.4	98.0			92.5
MID	106.5	110.3	102.6	94.8	92.7	89.0	94.0	
END	98.7			98.7	90.9			95.1

OFF-DESIGN CONDITIONS

(W.S. 156.83, WEIGHT = 13,154 Kg (29,000 LB)

ALT.	11,582M (38,000 FT)	10,363M (34,000 FT)	9,149M (30,000 FT)	11,582M (38,000 FT)	10,363M (34,000 FT)	9,144M (30,000 FT)
MACH						
.70	97.9			93.8		
.75	110.3	116.1	119.1	89.0	99.1	107.7
.77	108.3			95.7		

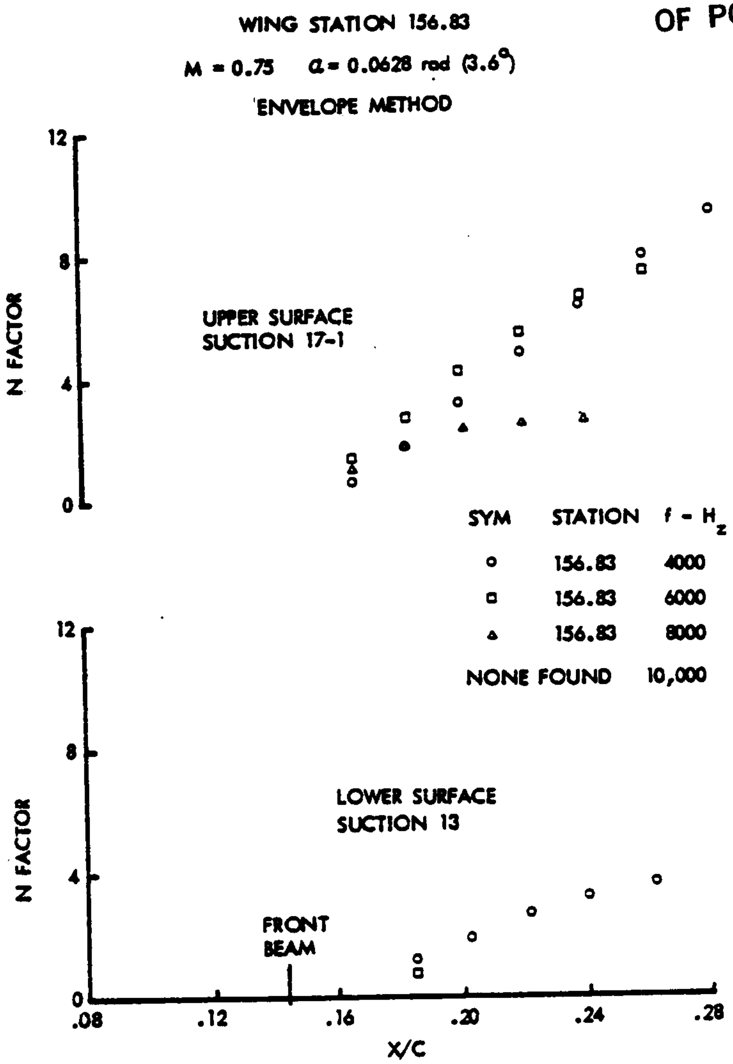


Figure 226. Mid-Cruise Tollmien-Schlicting N Factors

The suction levels and distribution indicated by the stability codes for the MOD8C airfoil were adjusted to produce an equivalent suction distribution compatible with physical limitations and the limitations imposed by fabrication techniques, slot flow criteria, and suction capability. This was of course an iterative process. Initial attempts at defining the required suction distribution involved delaying suction past 1 percent chord. This was desired because of the limited volume in the JetStar leading edge. A suction distribution was established starting aft of 1 percent. This distribution required high suction levels and chordwise extent of suction, and the capability of available suction pumps became a limiting factor. The suction distribution shown in Figure 227 was established to alleviate this problem while still satisfying the stability requirements. This suction distribution was established at the mid-cruise point: Mach No. = 0.75, altitude = 11,582 m (38,000 ft), and aircraft weight = 13,154 kg (29,000 lbs). Boundary layer stability was subsequently analyzed at other flight conditions.

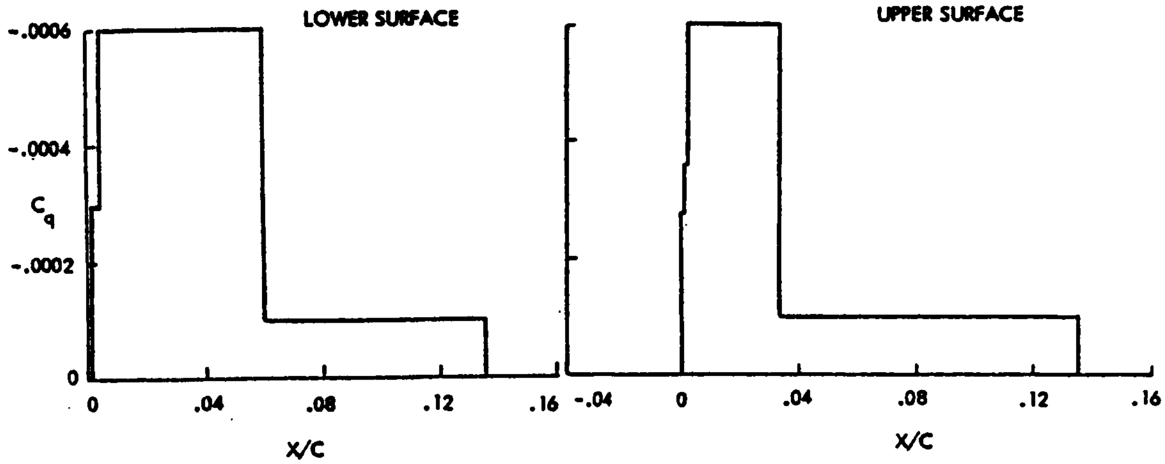


Figure 227. Required Suction Distributions

7.6.2 Suction Surface Design

The suction surface design of the leading-edge test article involved development of an array of discrete spanwise suction slots to provide the equivalent of the required distributed suction. The active leading-edge test surface extends from wing station 141.14 outboard to wing station 188.09 and aft to about 12 percent chord. The results were suction slots about 137.67 cm (54.2 in) in length. The surface design is based upon certain slot design criteria and physical limitations. These are illustrated in Figure 228. This figure also shows the relationship between slot Reynolds number and slot spacing for a specified suction level. The relationships among these parameters are illustrated graphically in Figure 229. All combinations of slot widths and slot Reynolds numbers which fall inside the shaded area of Figure 229 satisfy the slot criteria for a particular chordwise slot location and wing station. Additional envelopes can be computed for other wing locations. The ΔC_N curve would shift up or down for an increasing or decreasing distributed suction levels, respectively. These design envelopes were used to aid in development of the LETA suction surface.

7.6.2.1 Slot Design Criteria

The slot design criteria used to design and evaluate the test article suction surface are shown in Figure 228. The criteria limits were largely derived from NORAIR and X-21 sources. More recent Lockheed-funded tests have indicated that the limits shown for W/Z and U_z/U_e may be somewhat more restrictive than necessary, while the R_w limit may be excessive for the Lockheed slot/ducting configuration. The criteria limits were used as goals. In cases where it was difficult or impossible to satisfy the criteria, Lockheed tests were used to support relaxation of the limits. The specified physical limits represent current structural and fabrication limitations.

7.6.2.2 Spanwise C_p Variation

The slot parameters used as design criteria are computed from the predicted boundary layer velocity profiles, surface pressure and other fluid properties at the slot locations. Another important parameter with respect to suction surface

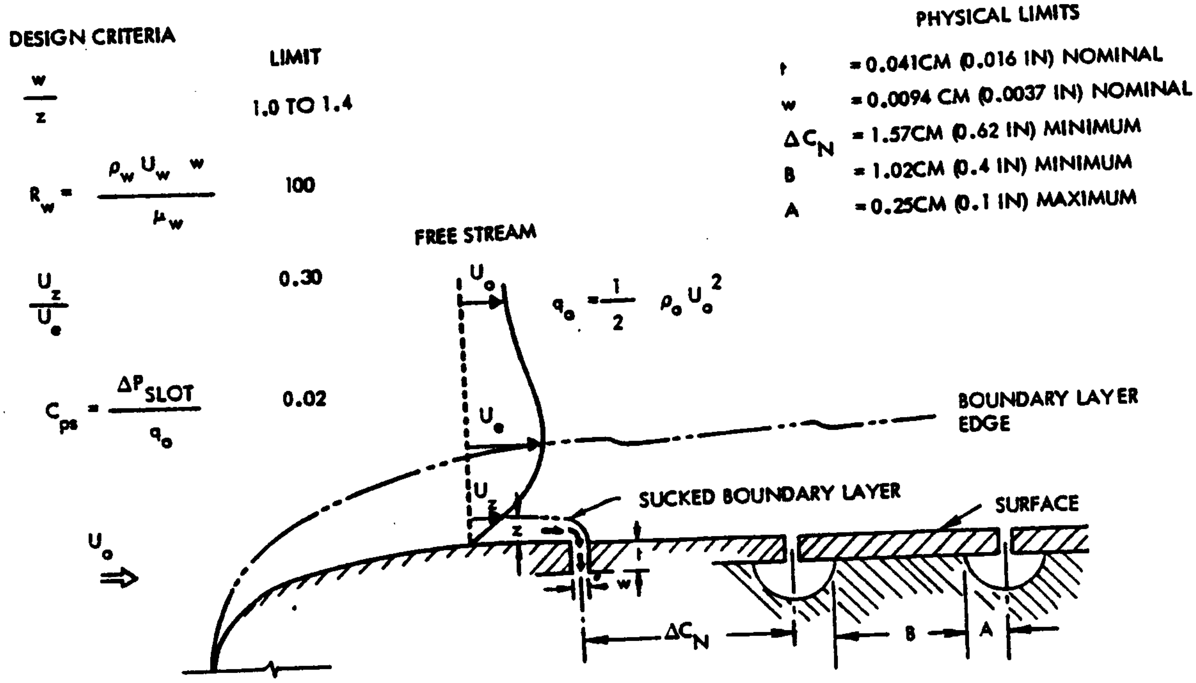


Figure 228. Slot Design Criteria and Physical Limitations

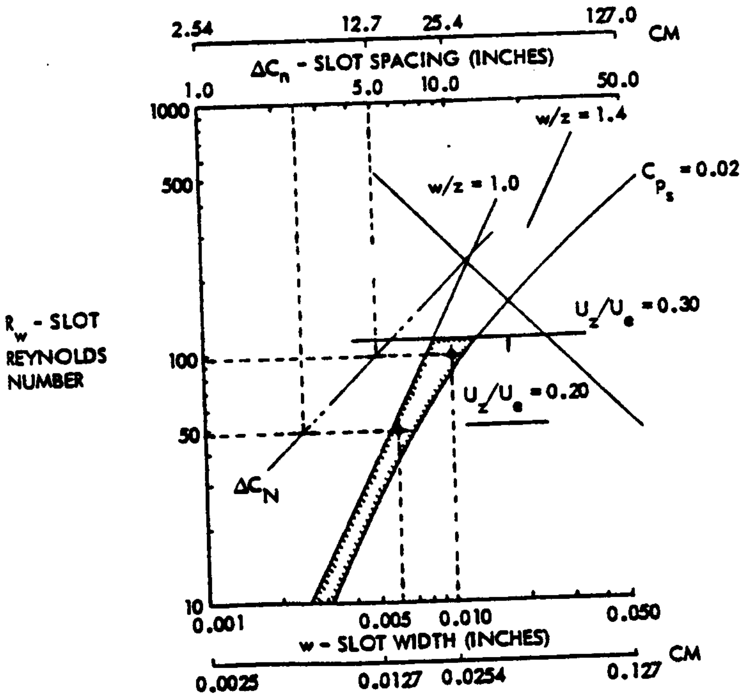


Figure 229. Sample Slot Design Envelope

performance is the spanwise surface pressure variation along a given slot. This factor did not directly impact the design of the suction surface, but its importance resulted in some redesign of the leading-edge airfoil. Attempts were made to minimize the spanwise Cp variation and to reduce it to acceptable levels while retaining other desired airfoil characteristics. The NORAIR criteria required a maximum allowable C_p variation of 0.0656 per meter (0.02 per foot) of slot. The pressure distributions for the final airfoil design are shown in Figure 230. Pressure profiles are shown for four wing stations, that include the inboard and outboard extremities of the active suction surface of the test article. Figure 230 clearly indicates that significant spanwise Cp variations still exist over much of the suction surface. Furthermore, the very steep slopes of these profiles over much of the test surface represent substantial changes in the surface pressures for relatively small errors in calculated Cp profile.

As mentioned, the Cp variation, while of much concern, did not impact the suction surface design. The slot development was based on suction surface analysis at selected discrete spanwise locations on the upper and lower test surfaces. Boundary layer parameters varied slightly from wing station to wing station, but the required suction distribution as shown in Figure 227 did not vary with spanwise location.

7.6.2.3 Slot Configuration

Previous NASA LFC contracted studies have indicated that the isobars on the wing of the study aircraft closely paralleled constant x/c elements. Based on these results, it had been planned to cut the suction slots of the LEFT test article along lines of constant x/c. However, by cutting the slots parallel to the leading edge of the test articles some relief is obtained from the undesirable spanwise Cp variation. Consequently, the surface design reflects slots cut in this manner.

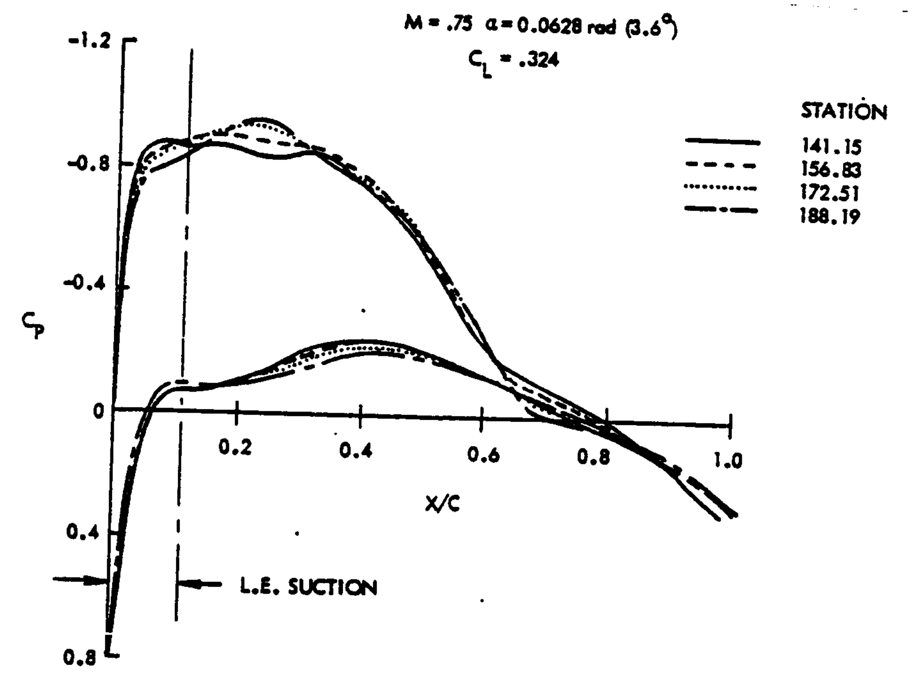


Figure 230. Mid-Cruise Pressure Distributions

The suction surface design resulted in the slot locations as shown in Figure 231. The design requires 12 slots on the upper surface and 13 slots on the lower surface. Since the suction region overlaps the region where cleaning/anti-icing slots are required, 1 upper-surface slot and 5 lower-surface slots serve a dual purpose. The slot locations and slot parameters for the inboard design point are presented in Tables 17 and 18.

The slot configuration was developed as a minimum slot configuration that satisfies the slot criteria within the established physical limits of the test article. Special attention was given to maintaining relatively low slot Reynolds numbers. Table 17 lists the upper-surface slot parameters at the inboard design point. The first slot was located as far forward as possible to minimize peak suction, while maintaining a minimum C_{ps} value of 0.02. The relatively low slot Reynolds numbers are favorable, and C_{ps} values are generally good. The W/Z and U_z/U_e in the early slots are undesirably high, but cannot be reduced because of physical limitations.

Table 18 lists the lower-surface slot parameters at the inboard design point. The relatively low slot Reynolds numbers and high C_{ps} values are favorable. W/Z values are satisfactory. The U_z/U_e are higher than desirable, but like the upper surface cannot be reduced because of physical limitations.

As noted earlier, Lockheed-funded tests have indicated that the U_z/U_e and W/Z criteria may be more restrictive than necessary. In light of these indications, the required suction distribution, and the unyielding physical limitations, the surface design is considered acceptable for the flight test program despite exceeding established criteria on some slots.

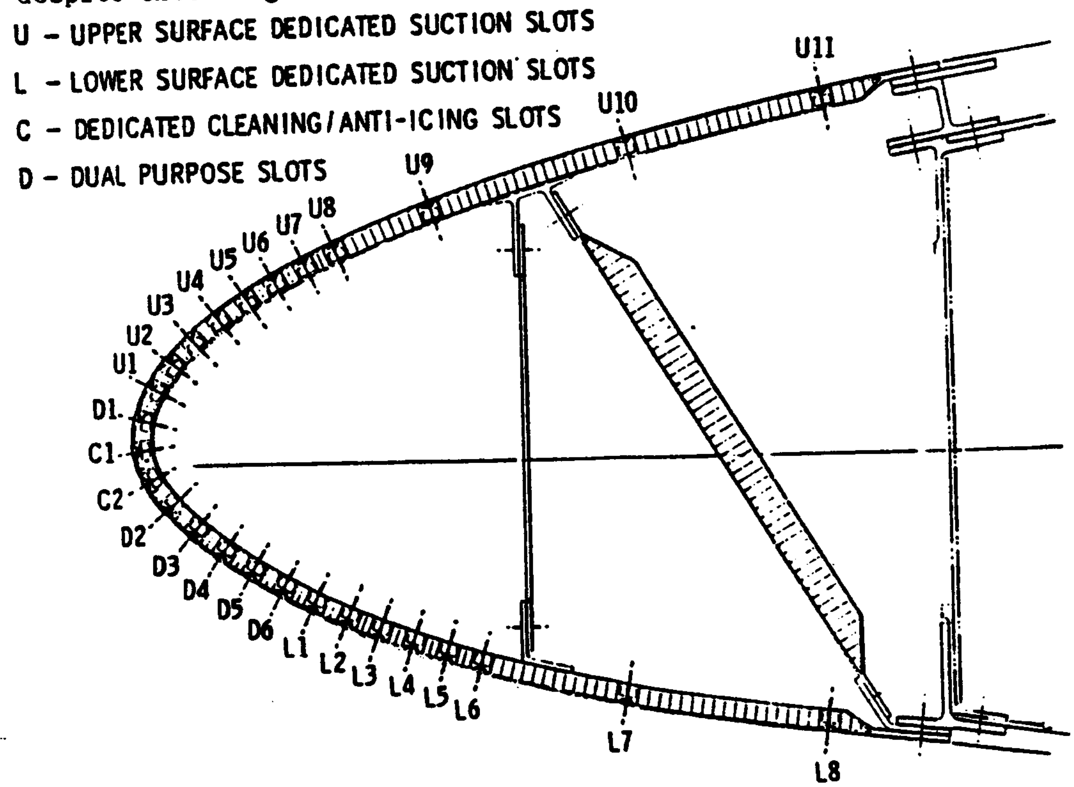


Figure 231. Leading Edge Slot Locations

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TABLE 17. UPPER SURFACE SLOT PARAMETERS

SLOT NO.	X/C	W CM	W IN	ΔC_N	R_W	W/Z	U_Z/U_e	C_{PS}
D1	.00055	.0094	.0037	0.83	31	2.27	0.48	0.025
U1	.0027	.0094	.0037	0.62	35	2.11	0.43	0.033
U2	.0058	.0094	.0037	0.62	45	1.74	0.43	0.053
U3	.0097	.0094	.0037	0.62	45	1.62	0.40	0.057
U4	.0140	.0094	.0037	0.62	45	1.50	0.37	0.061
U5	.0186	.0094	.0037	0.62	45	1.42	0.35	0.064
U6	.0234	.0094	.0037	0.62	45	1.35	0.34	0.066
U7	.0283	.0094	.0037	0.62	45	1.29	0.33	0.067
U8	.0333	.0094	.0037	0.62	44	1.24	0.31	0.067
U9	.0495	.0102	.0040	1.95	44	1.03	0.24	0.057
U10	.0811	.0114	.0045	3.70	44	0.97	0.20	0.043
U11	.1130	.0127	.0050	3.70	44	0.99	0.19	0.033

TABLE 18. LOWER SURFACE SLOT PARAMETERS

SLOT NO.	X/C	W CM	W IN	ΔC_N	R_W	W/Z	U_Z/U_e	C_{PS}
D2	.0052	.0094	.0037	0.60	30	1.28	0.74	0.021
D3	.0093	.0094	.0037	0.64	46	1.24	0.65	0.038
D4	.0138	.0094	.0037	0.64	46	1.32	0.55	0.040
D5	.0186	.0094	.0037	0.64	46	1.36	0.49	0.041
D6	.0236	.0094	.0037	0.64	46	1.36	0.45	0.042
L1	.0287	.0094	.0037	0.64	46	1.36	0.42	0.043
L2	.0340	.0094	.0037	0.64	46	1.35	0.40	0.044
L3	.0392	.0094	.0037	0.64	46	1.34	0.38	0.045
L4	.0447	.0094	.0037	0.64	46	1.32	0.37	0.046
L5	.0501	.0094	.0037	0.64	46	1.29	0.36	0.047
L6	.0556	.0094	.0037	0.64	46	1.26	0.34	0.048
L7	.0792	.0102	.0040	2.70	43	1.08	0.24	0.036
L8	.10711	.0114	.0045	3.10	44	0.99	0.21	0.029

7.6.3 Suction Ducting Design

The required suction distribution and slot locations substantially define the suction flows for each slot line. Unfortunately, since a significant Cp variation is still predicted along many slots, some excess suction must occur over portion of all slots to ensure achievement of the design suction level at all points along a slot. The slot metering systems are designed to minimize their sensitivity to spanwise Cp variations. Nonetheless, the system does not provide for control of spanwise flow variations, and some excess flow is predicted for all slots. But some relief is possible. The inherent oversuction can be minimized by applying suction at the appropriate end of the manifold or piccolo tube of each slot. By applying suction at the tube end that corresponds

to the lowest surface pressure, the oversuction will be less than if suction were applied to the opposite tube end. This has been done for all slots except U2, L1, L2, L3, L7, and L8. Space limitations inside the test article prevent plumbing all slots in the desired manner.

The design flow for the suction system ducting is based upon 150 percent of the design suction distribution as defined in Section 7.1. The ducting system design includes all suction lines and components between the LETA interface and the inlet to the sonic needle valves of each chamber valve. The design flow parameters for the Lockheed system are given in Table 19. These parameters result from the distributed suction requirements, the suction surface design, the LETA Cp distribution, and pressure losses inside the LETA. The equivalent McDonnell Douglas system flows and pressures were provided by McDonnell Douglas as shown in Table 20.

The ducting system has been developed to maximize the total pressure delivered to the inlet of the sonic needle valves at the design flows. The limiting factor is the space available for suction tubes in the JetStar leading edges. A schematic diagram of the ducting system illustrating the primary components and sources of pressure loss is presented in Figure 232. A computer code was developed to model this configuration and predict the pressure losses. The results of this design development are given in Table 21 for the Lockheed system and in Table 22 for the McDonnell Douglas system. The line identifications in these tables are keyed to Figure 232. Some disparity will be noted in the needle valve inlet total pressures given in Tables 21 and 22. Ideally, this pressure would appear as a single value for each table, maximized within the physical limits of the aircraft. The disparity results from the requirement and the desire to only specify standard and readily obtainable line sizes. These discrete line sizes dictate discrete needle valve pressures and preclude an exact match of these pressures.

The leading-edge line of McDonnell Douglas flute Number 1 was originally sized at 1.588 cm (0.625 in) OD but 1.905 cm (0.750 in) is specified, since this avoids a transition piece to match McDonnell Douglas' adapter and the small additional space was available. This accounts for the relatively large needle-valve pressure of flute Number 1. In general, attempts were made to maintain a maximum duct Mach number of about 0.20 for suction lines in the leading edge and the fuselage. The primary purpose was to keep internal flow disturbances, and thus internal noise, as low as practical. For the design flow case, the maximum Mach numbers predicted for the Lockheed system are 0.26 and 0.24 for the leading-edge lines and the fuselage lines, respectively. The equivalent values for the McDonnell Douglas system are 0.17 and 0.16.

In all cases where a suction line size transition is required at the LETA interface, low noise adapters are specified. These adapters are simple conical diffusers. Their total included diffusion angle is about 0.10-0.14 rad (6 to 8 degrees), space permitting, to minimize internal flow disturbances in the transition to a larger line size.

All lines were sized assuming that a Hastings flowmeter is installed in the fuselage line (line Number 2). This line sizing considered total pressure losses and plumbing adapters to fit the flowmeters. In general, flows will be measured by means of a NASA-furnished flow calibration of the sonic needle

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TABLE 19. LOCKHEED SUCTION SYSTEM DUCTING DESIGN PARAMETERS

UPPER SURFACE			LOWER SURFACE		
SLOT NUMBER	MASS FLOW (KG/SEC)	LINE PRESSURE AT INTERFACE (KPa)	SLOT NUMBER	MASS FLOW (KG/SEC)	LINE PRESSURE AT INTERFACE (KPa)
D1	0.001085	18.86	D2	0.001115	13.41
U1	0.001283	14.65	D3	0.001580	18.48
U2	0.001507	14.84	D4	0.001605	17.00
U3	0.001458	12.83	D5	0.001622	15.66
U4	0.001486	11.44	D6	0.001624	14.56
U5	0.001498	10.58	L1	0.001717	17.19
U6	0.001504	9.72	L2	0.001705	16.57
U7	0.001516	9.43	L3	0.001688	16.09
U8	0.001500	8.86	L4	0.001640	11.59
U9	0.001509	8.52	L5	0.001619	12.21
U10	0.001493	8.67	L6	0.001604	11.73
U11	0.001428	9.77	L7	0.001466	15.90
			L8	0.001486	15.94
TOTAL	0.017281			0.020457	

NOTE: FLOW TEMPERATURE AT INTERFACE = -32°C

UPPER SURFACE			LOWER SURFACE		
SLOT NO.	MASS FLOW (LB/SEC)	LINE PRESSURE AT INTERFACE (PSFA)	SLOT NO.	MASS FLOW (LB/SEC)	LINE PRESSURE AT INTERFACE (PSFA)
D1	.002392	394	D2	.002458	280
U1	.002828	306	D3	.003484	386
U2	.003323	310	D4	.003539	355
U3	.003214	268	D5	.003575	327
U4	.003275	239	D6	.003581	304
U5	.003302	221	L1	.003786	359
U6	.003316	203	L2	.003759	346
U7	.003342	197	L3	.003721	336
U8	.003308	185	L4	.003616	242
U9	.003327	178	L5	.003570	255
U10	.003292	181	L6	.003537	245
U11	.003148	204	L7	.003231	332
			L8	.003276	333
TOTAL	0.0381			0.0451	

NOTE: FLOW TEMPERATURE AT INTERFACE = -26°F

TABLE 20. McDONNELL DOUGLAS SUCTION SYSTEM DUCTING DESIGN PARAMETERS

FLUTE NUMBER	MASS FLOW (KG/SEC)	LINE PRESSURE AT INTERFACE (KPa)
1	0.003057	25.62
2	0.005225	20.54
3	0.004450	16.76
4	0.003252	14.99
5	0.003361	13.50
6	0.003715	12.83
7	0.003184	12.59
8	0.002028	12.78
9	0.001560	12.88
10	0.001606	12.74
11	0.001751	12.64
12	0.001751	12.69
13	0.001751	12.64
14	0.001751	12.69
15	0.001751	12.69
TOTAL	0.040189	

FLUTE NO.	MASS FLOW (LB/SEC)	LINE PRESSURE AT INTERFACE (PSFA)
1	.00674	535
2	.01152	429
3	.00981	350
4	.00717	313
5	.00741	282
6	.00819	268
7	.00702	263
8	.00447	267
9	.00344	269
10	.00354	266
11	.00386	264
12	.00386	265
13	.00386	264
14	.00386	265
15	.00386	265
TOTAL	0.0886	

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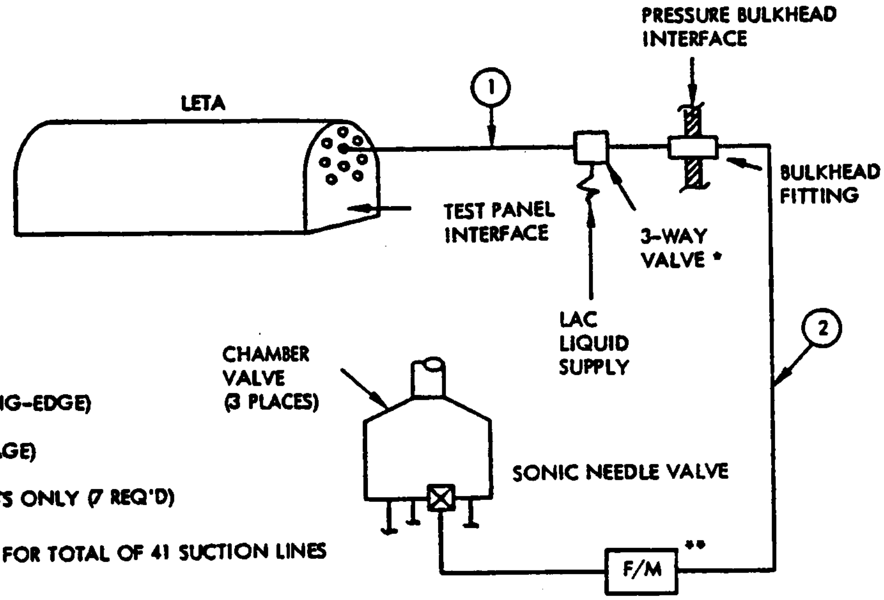

(1) SLOT LINE NO. 1 (LEADING-EDGE)

(2) SLOT LINE NO. 2 (FUSELAGE)

*DUAL-PURPOSE OR CLN/AI SLOTS ONLY (7 REQ'D) (LOCKHEED SYSTEM ONLY)

**MAXIMUM OF SIX FLOWMETERS FOR TOTAL OF 41 SUCTION LINES

Figure 232. Ducting System Schematic

valves. The flowmeters will provide a check on selected needle valve flow indications. They will also provide a backup in the event that some needle valves must operate unchoked under certain flight and suction conditions.

7.7 AIRCRAFT PERFORMANCE

Take-off and landing performance calculations were made for the LFC JetStar. Two configuration changes and a change in operating technique are the major factors influencing the performance calculations. Specifically, these are:

(1) Removal of the external fuel tanks.

(2) Locking the leading-edge slats in the retracted position.

(3) Increased take-off and landing speeds.

A full report of the LFC JetStar performance may be found in Reference 8.

7.7.1 Cruise Performance

Because replacement of the external fuel tanks with the LFC glove will result in a decrease in cruise drag, cruise performance for the modified aircraft was not calculated. Use of existing cruise data will provide a slight conservatism for cruise performance.

7.7.2 Takeoff and Landing Performance

The out-of-ground effect lift and drag data were based on the Calspan wind-tunnel test. Remaining data were taken from the 1964 JetStar Flight Manual and the Aerodynamic Substantiating Data Reports, References 9 and 10.

TABLE 21. LINE SIZES AND NEEDLE VALVE PRESSURE-LOCKHEED

LOCKHEED UPPER SURFACE			
SLOT NUMBER	LINE 1 (O.D.-CM)	LINE 2 (O.D.-CM)	P_N (KPa abS)
C1/C2	1.588	1.588	-
D1	1.270	1.270	9.337
U1	1.588	1.588	10.821
U2	1.588	1.588	9.337
U3	1.588	1.905	8.283
U4	1.588	3.810	6.847
U5	1.905	1.905	7.134
U6	1.905	3.810	7.182
U7	1.905	3.810	6.703
U8	2.223	2.223	6.512
U9	2.223	2.540	6.368
U10	2.223	2.540	6.607
U11	1.905	2.540	7.278
LOCKHEED LOWER SURFACE			
D2	1.588	1.588	9.385
D3	1.588	1.588	12.880
D4	1.588	1.588	10.390
D5	1.588	1.588	7.757
D6	1.588	1.905	7.469
L1	1.588	1.588	11.683
L2	1.588	1.588	10.869
L3	1.588	1.588	10.246
L4	1.905	1.905	8.140
L5	1.588	3.810	7.182
L6	1.588	3.810	6.416
L7	1.588	1.588	11.635
L8	1.588	1.588	11.587

LOCKHEED UPPER SURFACE			
SLOT NUMBER	LINE 1 (O.D.-IN)	LINE 2 (O.D.-IN)	P_N (LB/FT^2 abS)
C1/C2	0.625	0.625	-
D1	0.500	0.500	195
U1	0.625	0.625	226
U2	0.625	0.625	195
U3	0.625	0.750	173
U4	0.625	1.500	143
U5	0.750	0.750	149
U6	0.750	1.500	150
U7	0.750	1.500	140
U8	0.875	0.875	136
U9	0.875	1.000	133
U10	0.875	1.000	138
U11	0.750	1.000	152
LOCKHEED LOWER SURFACE			
D2	0.625	0.625	196
D3	0.625	0.625	269
D4	0.625	0.625	217
D5	0.625	0.625	162
D6	0.625	0.750	156
L1	0.625	0.625	244
L2	0.625	0.625	227
L3	0.625	0.625	214
L4	0.750	0.750	170
L5	0.625	1.500	150
L6	0.625	1.500	134
L7	0.625	0.625	243
L8	0.625	0.625	242

NOTE: P_N IS TOTAL PRESSURE AT INLET TO SONIC NEEDLE VALVE

TABLE 22. LINE SIZES AND NEEDLE VALVE PRESSURE-McDONNELL DOUGLAS

FLUTE NUMBER	LINE 1 (O.D.-CM)	LINE 2 (O.D.-CM)	P_N (KPa abS)
1	1.905	1.905	20.780
2	2.540	2.540	11.970
3	2.540	3.175	10.007
4	2.540	2.540	10.534
5	2.540	3.175	8.762
6	3.810	3.810	8.379
7	3.175	3.175	8.906
8	1.905	2.540	9.097
9	1.905	1.905	10.103
10	1.905	1.905	9.768
11	1.905	1.905	9.049
12	1.905	1.905	9.097
13	1.905	1.905	9.049
14	1.905	1.905	9.097
15	1.905	1.905	9.097
FLUTE NUMBER	LINE 1 (O.D.-IN)	LINE 2 (O.D.-IN)	P_N (LB/FT^2 abS)
1	0.750	0.750	434
2	1.000	1.000	250
3	1.000	1.250	209
4	1.000	1.000	220
5	1.000	1.250	183
6	1.500	1.500	175
7	1.250	1.250	186
8	0.750	1.000	190
9	0.750	0.750	211
10	0.750	0.750	204
11	0.750	0.750	189
12	0.750	0.750	190
13	0.750	0.750	189
14	0.750	0.750	190
15	0.750	0.750	190

NOTE: P_N IS TOTAL PRESSURE AT INLET TO SONIC NEEDLE VALVE

Take-off and landing speeds are increased above those of the basic JetStar Flight Manual and the Aerodynamic Substantiating Data Reports, References 9 and 10.

Take-off and landing speeds are increased above those of the basic JetStar to simulate the speeds of a large transport. This will allow a representative simulation and evaluation of the leading-edge washing system. These speeds are below any limiting speeds for altitudes up to 1,219 m (4,000 ft) and temperatures up to 25°C (77°F) above standard day. Also, these speeds allow at least a 30 percent stall margin for the normal flap settings, even though stall speeds are slightly higher than those of the basic JetStar.

Take-off and landing field lengths (including the 1/0.6 factor) do not exceed 2,743 m (9,000 ft), within the altitude and temperature limitations previously mentioned. Summaries of the take-off and landing data are shown in Tables 23 and 24 respectively.

TABLE 23. TAKEOFF DATA SUMMARY

GROSS WEIGHT 1000 KG	V_1	V_R	V_2	TOFL (M)	V_1	V_R	V_2	TOFL (M)	V_1	V_R	V_2	TOFL (M)
	S.L. (15°C)				610 M (11°C)				1,219 M (7.1°C)			
15.4	61.2	72.0	76.7	1,661	62.8	72.5	76.7	1,875	63.8	73.6	76.7	2,137
14.5	60.2	72.0	76.7	1,548	61.7	72.5	76.7	1,750	62.8	73.6	76.7	1,990
13.6	59.2	71.5	76.1	1,436	60.2	72.5	76.7	1,625	61.7	73.6	76.7	1,847
12.7	57.6	71.5	76.1	1,329	59.2	72.5	76.7	1,500	60.2	73.6	76.7	1,704
11.8	56.1	71.0	76.1	1,225	56.1	72.0	76.7	1,381	59.2	73.1	76.7	1,570
10.9	55.0	70.5	76.1	1,128	55.6	71.5	76.1	1,268	56.1	73.1	76.7	1,469
	S.L. (30°C)				610 M (26°C)				1,219 M (22.1°C)			
15.4	62.8	72.0	76.1	1,899	63.8	73.6	76.7	2,146	64.8	74.1	76.7	2,432
14.5	61.7	72.0	76.1	1,768	62.8	73.1	76.1	2,000	63.8	74.1	76.7	2,262
13.6	60.7	72.0	76.1	1,640	61.7	73.1	76.1	1,853	62.8	73.6	76.7	2,094
12.7	59.2	72.0	76.1	1,823	60.7	73.1	76.1	1,713	61.7	73.6	76.7	1,939
11.8	57.6	72.0	76.1	1,396	59.2	73.1	76.1	1,573	60.2	73.6	76.7	1,789
10.9	56.1	71.5	76.1	1,280	57.6	72.5	76.1	1,442	58.6	73.6	76.7	1,643
	S.L. (40°C)				610 M (36°C)				1,219 M (32.1°C)			
15.4	64.3	73.6	76.1	2,100	64.8	74.1	76.1	2,365	66.4	74.6	76.7	2,694
14.5	62.8	73.6	76.1	1,954	63.8	74.1	76.1	2,207	65.3	74.6	76.7	2,509
13.6	61.7	73.1	76.1	1,814	62.8	73.6	76.1	2,045	64.3	74.6	76.7	2,329
12.7	60.7	73.1	76.1	1,676	61.7	73.6	76.1	1,890	63.3	74.6	76.7	2,155
11.8	59.2	72.5	76.1	1,539	60.2	73.6	76.1	1,737	61.7	74.1	76.7	1,984
10.9	57.6	72.5	76.1	1,411	58.6	73.1	76.1	1,591	60.2	74.1	76.7	1,817

GROSS WEIGHT 1000 LB	V_1	V_R	V_2	TOFL (FT)	V_1	V_R	V_2	TOFL (FT)	V_1	V_R	V_2	TOFL (FT)
	S.L. (59°F)				2000 FT (52°F)				4000 FT (45°F)			
34	119	140	149	5450	122	141	149	6150	124	143	149	7010
32	117	140	149	5080	120	141	149	5740	122	143	149	6530
30	115	139	148	4710	117	141	149	5330	120	143	149	6060
28	112	139	148	4360	115	141	149	4920	117	143	149	5590
26	109	138	148	4020	112	140	149	4530	115	142	149	5150
24	107	137	148	3700	108	139	148	4160	112	142	149	4820
	S.L. (77°F)				2000 FT (79°F)				4000 FT (72°F)			
34	122	140	148	6230	124	143	149	7040	126	144	149	7980
32	120	140	148	5800	122	142	148	6560	124	144	149	7420
30	118	140	148	5380	120	142	148	6080	122	143	149	6870
28	115	140	148	5980	118	142	148	5620	120	143	149	6360
26	112	140	148	4580	115	142	148	5160	117	143	149	5870
24	109	139	148	4200	112	141	148	4730	114	143	149	5390
	S.L. (104°F)				2000 FT (97°F)				4000 FT (90°F)			
34	125	143	148	6890	126	144	148	7760	129	145	149	8840
32	122	143	148	6410	124	144	148	7240	127	145	149	8230
30	120	142	148	5950	122	143	148	6710	125	145	149	7640
28	118	142	148	5500	120	143	148	6200	123	145	149	7070
26	115	141	148	5050	117	143	148	5700	120	144	149	6510
24	112	141	148	4630	114	142	148	5220	117	144	149	5960

NOTE: (1) SLATS RETRACTED
(2) ALL SPEEDS KEAS
(3) 4-ENGINE V_2 = 155 KEAS FOR ALL WEIGHTS AND CONDITIONS
(4) TRAILING EDGE FLAP DEFLECTION, $\delta_F = 20°$

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TABLE 24. LANDING DATA SUMMARY

GROSS WEIGHT		LANDING FLAP V_{APP}		LANDING FLAP V_{STALL}		CLEAN V_{STALL}	
1,000 KG	1,000 LB	M/SEC	KEAS	M/SEC	KEAS	M/SEC	KEAS
15.4	34	74.6	145	57.1	111	61.7	120
14.5	32	74.6	145	55.0	107	59.7	116
13.6	30	74.6	145	53.5	104	58.1	113
12.7	28	74.6	145	51.4	100	56.1	109
11.8	26	74.6	145	49.9	97	54.0	105
10.9	24	74.6	145	47.8	93	52.0	101

NOTE: (1) SLATS RETRACTED

(2) TRAILING EDGE LANDING FLAP DEFLECTION, δ_F = 0.873 rad (50 DEG)

8.0 STRUCTURAL CRITERIA AND STRESS ANALYSIS

8.1 STRUCTURAL CRITERIA

This section summarizes the structural criteria used for design of the LFC-LEFT modifications to the JetStar aircraft. Results of the CALSPAN 0.10 scale glove model wind-tunnel test did not indicate that any modification to the analytical wing chordwise or leading edge shield pressure distributions was required. However, the wind-tunnel test did indicate that some adjustments to the aircraft wing aerodynamic forces and moments were necessary. These adjustments were made primarily due to the differences in analytical prediction of wing airloading with and without the external fuel tanks.

8.1.1 Structural Design and Modification Requirements

The structural design and modification requirements presented in this section are applicable to the LFC leading edge glove modification of the JetStar aircraft new or modified structure only. The loads, load factors, and pressures presented in this section are LIMIT values unless otherwise specified.

8.1.2 External Load Requirements

The following external load criteria were applied:

(a) The structure shall be capable of withstanding limit loads without suffering detrimental deformations that would interfere with the safe operation of the aircraft.

(b) Except in certain cases where additional strength or multiplying factors of safety are specified, a minimum factor of safety of 2.0 shall apply to all limit load conditions for new or modified structure only. The structure shall be capable of withstanding ultimate load without failure.

8.1.3 Internal Load Requirements

The following internal load criteria were applied:

(a) Casting factors, bearing factors, and fitting factors shall be consistent with values previously established by CAR4b.

(b) Allowable static strength data shall be taken whenever possible and in order of precedence from the MIL-HDBK-5, MIL-HDBK-23, MIL-HDBK-17, or other applicable NASA and National Bureau of Standards documents.

(c) Provisions shall be made for temperature effects associated with the anticipated hot and cold atmospheres and systems operations.

(d) Selection of the physical properties used in the structural design shall include a consideration of all factors that affect the allowable strength.

8.1.4 Fail-Safe Requirements

The fail-safe criteria were:

(a) The design of the new or modified structure shall be such that catastrophic failure or excessive structural deformation, which would cause loss of control of the aircraft, are not probable after fatigue failure or obvious partial failure of a single principal structural element. After such failure, the remaining structure shall be capable of withstanding the limit design flight loads of this report.

(b) The flap shall be designed fail-safe for the failure of any axial member, shear panel, either skin of any honeycomb panel, or any attach fitting.

8.1.5 Maximum Strains at 12 percent Chord Interface

The maximum ultimate strains along the 12 percent chord interface with the leading-edge test panel were applied. These are given below with both positive and negative values indicating tension and compression, respectively. The strains are based on previous design conditions for the wing with consideration given for the location of the interfaces.

INTERFACE	POSITIVE STRAIN CM/CM (IN/IN)	NEGATIVE STRAIN CM/CM (IN/IN)
Upper	0.0015	0.0042
Lower	0.0021	0.0018

8.1.6 Attachment of Ribs to Front Spar

The following criteria were used for rib attachment:

(a) Attachment of leading-edge ribs to the wing front spar should be accomplished in a manner to distribute chordwise loads as near as possible to the wing upper and lower surface panels.

(b) Vertical shear loads shall be distributed into the spar web with appropriate shear ties.

8.1.7 Leading-Edge Shield Requirements

The following criteria applied in design of the McDonnell-Douglas shield:

(a) The leading-edge shield shall be controlled by two actuators to provide a fail-safe design in the event one actuator, its support structure, or actuation mechanism fails. In the event of a failure of the drive system, including torque tubes, the actuators shall be irreversible under limit loads on the flap.

(b) The leading-edge shield system shall incorporate mechanical stops to stall the drive system in both the extend and retract directions.

These stops shall be in addition to the electrical limit switches and any secondary stop provided by a mechanically controlled hydraulic shut-off valve.

(c) The leading-edge shield shall be designed to meet the requirements of CAR4b.310 and 4b.220(e).

8.1.8 Flutter Design Requirements

The leading edge was designed to comply with the flutter requirements of FAR25.629.

(a) The leading edge shall be designed to be free from flutter, divergence and excessive vibration for all points of the flight envelope up to 1.2 V_D/M_D for the retracted leading-edge shield and up to 120 percent of the maximum test speed (V = 128.6 m/sec (250 KEAS)/M = 0.4) for the extended leading-edge shield.

(b) The design shall be free from flutter and divergence at any speed up to V_D/M_D for the retracted leading-edge shield and up to the maximum test speed (V = 128.6 m/sec (250 KEAS/M = 0.4)) for the extended leading-edge shield after the failure of any single structural member.

(c) The design shall meet the free play requirements of MIL-A-8870, "Airplane Strength and Rigidity - Vibration, Flutter and Divergence."

(d) To prevent any significant inertia coupling between the leading-edge shield vibration modes and the aircraft vibration modes, the leading-edge shield vibration frequencies should be at least 25 Hz for the unfailed condition.

8.1.9 Sonic Fatigue and Vibration

The sonic fatigue requirements of MIL-A-8893, "Airplane Strength and Rigidity - Sonic Fatigue," were followed in the leading-edge design.

8.1.9.1 Sonic Fatigue Design Criteria

The leading-edge sonic fatigue design shall be representative of a "1990's" subsonic transport aircraft equipped with four aft-mounted fanjet engines and having a mean design service life of 180,000 hours. The leading edge shall be designed for a JetStar mean service life of 3,000 hours. The JetStar service shall be presumed to include 1,350 hours at cruise with no laminar flow control, 1,350 hours at cruise with laminar flow control, 100 hours at maximum takeoff power, and 10 hours at maximum reverse thrust, wherein the rms pressures on the leading-edge that the structure shall withstand without failure or malfunction are as follows:

Octave Band Center Hertz	Cruise No LFC dB	Takeoff dB	Reverse Thrust dB
63	110	116	131
125	113	120	135
250	116	124	139
500	119	129	144
1000	122	130	145

8.2 JETSTAR STRESS ANALYSIS

The external design loads, flutter, sonic and failsafe requirements for the JetStar LFC modification were reported to NASA Langley as unpublished data titled "Preliminary Structural Requirements for JetStar LFC Leading Edge Glove Modification," SRD 72-73-843, Revision C, 14 September 1981.

The stress analysis accomplished to substantiate the static strength of the NASA JetStar for the LFC modification was also reported to NASA Langley as unpublished data in the form of a stress analysis report with modified aircraft working data.

8.2.1 Test Articles Description

The JetStar LFC modification consists of an installation of two different wing leading-edge test sections. The left side installation incorporates the Lockheed concept, and the right side incorporates the Douglas concept. The test sections consist of leading-edge gloves extending to the rear beam on the upper surface (approximately 65 percent chord) and to approximately 25 percent chord on the lower surface. The glove is approximately 1.83 m (6 feet) long and spans from wing station 134.75 to wing station 205.78. The fiberglass fairings form the transition from the test section to the wing surfaces. Lockheed was responsible for the design of the left test section, the attachment of both sections and fairings to the wing and the installation of the systems required to maintain the suction, anti-icing and cleaning system. The right wing test section was designed by McDonnell-Douglas Corporation.

8.2.2 JetStar Modifications

The test installation required the removal of the external fuel tanks. As a result, the following modifications were required:

(a) Extension of the trailing edge and flap from flap station 58.22 to flap station 97.03

(b) Relocation of the landing light from the nose of the external fuel tank to trailing edge lower surface at wing station 156.24.

8.2.3 JetStar Systems Additions

In addition to the above, the following system additions were required:

(a) Cleaning System - This consists of two tanks each filled with 60.57 liters (16 gallons) of PGME. The tanks are aft of the pressure

bulkhead and attached to the fuselage frames at fuselage stations 698.59, 711.50, and 725.50.

(b) Anti-Icing System (TKS LTD.) - This is required to de-ice the right wing shield. The system consists of a 13.25 liter (3.5 gallon) tank filled with ethyl glycol. The tank is located at the seat tracks between fuselage stations 459 and 468.

(c) Chamber Valve Installation - This controls the suction in the system. Two chamber valves are on the left side and one on the right side, located at the seat track locations. The left-side installations are located between fuselage stations 431 and 445.73, and between fuselage stations 455.96 and 470.69. The right-side installation is located between fuselage stations 455.96 and 470.69.

(d) Suction Pump Installation - The suction pump is required to maintain a pressure differential at the test section. The pump is located at fuselage station 608 aft of the JetStar aft pressure bulkhead.

8.2.4 Stress Analysis Results

The structural criteria to meet the static strength requirements are shown in Section 8.1 and were used in the analysis. The ultimate loads for the added structure are 2.0 x limit loads. For the unmodified structure, the ultimate loads were 1.5 x limit loads. Where appropriate, the structure was shown to be adequate by comparison with the loads used in the analysis of existing structure.

8.3 LEFT ARTICLE STRUCTURAL ANALYSIS

8.3.1 Structural Materials

The major materials of the LEFT assembly are listed in Figure 233. The outer skin is formed of 0.041 cm (0.016 in) 6A1-4V titanium. The load-carrying face sheets are HMF 343.34 graphite fabric with a thickness of 0.020 cm (0.008 in) per ply. The honeycomb core is Nomex-nylon.

The adhesives used for panel assembly are shown in Figure 234.

Selected tube materials are shown in Figure 235. The manifold tubes are made of 2024-T3 alloy for ease of forming. The piccolo tubes are made of 6061-T6 alloy for weldability. The flexible tube meets the requirements of MIL-H-8794 or MIL-H-5593.

Figure 236 lists the materials selected for diaphragms and attaching angles. The forward diaphragm web material is 2024-T3 clad sheet with a thickness of 0.318 cm (0.125 in). The 2024 material was selected on the basis of low cost and compatibility with access holes and installation in short lengths. The forward and aft diaphragm lower attach angles are made of 301 quarter-hard stainless steel. The upper fitting is made of graphite fabric. The aft diaphragm is made of graphite-fabric face sheets and a Nomex core.

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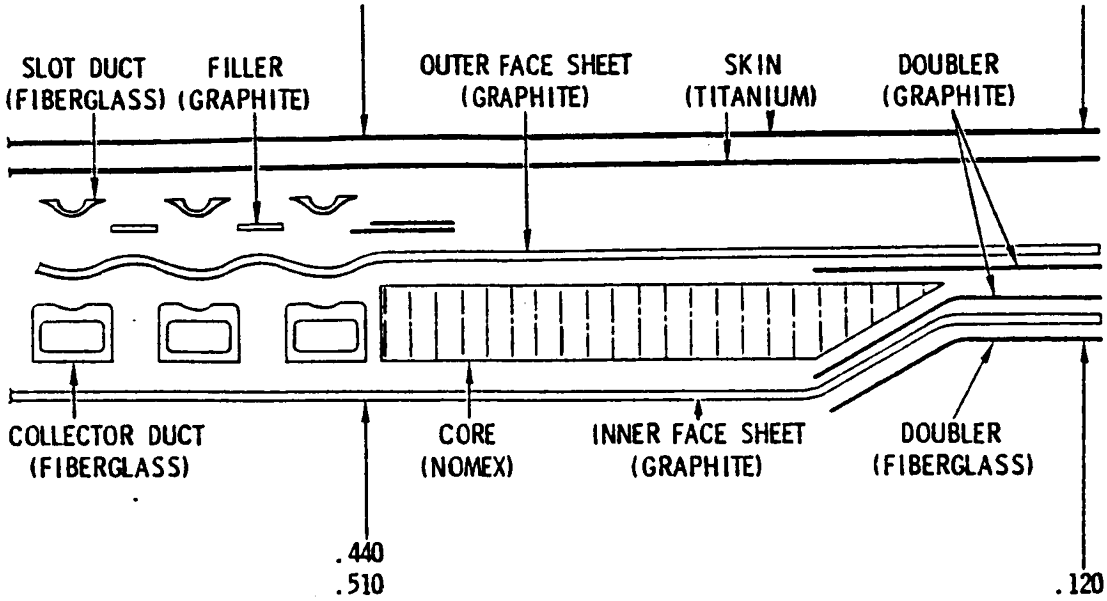

Figure 233. LEFT Assembly Major Materials

8.3.2 Thermal Stress Analysis

The following coefficients of thermal expansion were used in the analysis:

Wing (7075-T6 extrusion): 2322 x 10^{-6} °C (12.9x10^{-6} °F) (MIL Handbook 5)

Leading Edge (Graphite Fabric): 2.124x10^{-6} °C (1.18x10^{-6} °F) (SMN 400)

The spanwise thermal contraction for the cold condition along the 12 percent x/c front beam intersection was calculated to be 0.160 cm (0.063 in). To keep thermal loads from building up, the hole tolerance in the wing structure is equal to 0.160 cm (0.063 in.) for a 0.476 cm (0.188 in.) attach pin, the holes have a minimum diameter of 0.643 cm (0.253 in.)

The maximum vertical thermal expansion at the wing front beam was calculated to be 0.051 cm (0.020 in.) The maximum interference load for this thermal expansion is approximately 2.27 kg (5 lb) limit load.

In the internal loads section, the vertical thermal expansion internal loads are combined with the critical air loads and force bending internal loads.

8.3.3 Force Bending Analysis

Using the wing finite-element model, the vertical deflection of the front spar was calculated for the maximum up-bending condition. The maximum deflection from a straight line through the ends of the test section was calculated to be 0.330 cm (0.13 in.). By dividing this deflection by two, the maximum force deflection of 0.165 cm (0.065 in.) is obtained.

DRAWING	COMPONENT	MATERIAL ALLOY	FORM	REASON FOR SELECTION
LFC-18	PANEL ASSEMBLY			
	OUTER SKIN	6AL-4V TI	ANNEAL SHEET	CORROSION RESISTANCE + FORMABILITY
	SLOT DUCT	FIBERGLASS	PREPREG	EASY TO FORM
	FACE SHEETS (GR/EP)	HMF 343/34 (FIBERITE)	FABRIC (PREPREG)	FORMABILITY
	CORE	NOMEX-NYLON	10.16cm-0.318 x 0.914cm (4.0-$\frac{1}{8}$ x .36)	COMPATIBILITY WITH G/E
	COLLECTOR DUCTS	FIBERGLASS	PULTRUDED	LOW COST

DRAWING	COMPONENT	MATERIAL ALLOY	FORM	REASON FOR SELECTION
LFC-18	PANEL ASSEMBLY			
	BOND			
	TIT/TIT	EA9628	.030-OST	OST TO PERMIT SLIP INTO MOLD
	TIT/G/E	FM123-2	0.002 KPa (.045 PSF)	LOW FLOW
	SLOT DUCT/G/E	FM400	0.0034 KPa (.070 PSF)	HIGH TEMPERATURE
	G/E/COLLECTOR DUCT	EA 934	PUTTY	VOID FILLER, ROOM TEMPERATURE WITH POST CURE
	G/E /CORE	FM 400	0.0034 KPa (.070 PSF)	HIGH TEMPERATURE
	CORE SPLICE	AF3015/FM37	FOAM ADH.	HIGH TEMPERATURE

Figure 234. Panel Assembly Adhesives

It was noted that the aft diaphragm would have to transmit all of the vertical loads from the wing into leading edge. The aft-diaphragm attaching-pin-to-hole free-play tolerance exceeds the wing force bending deflection by a margin of 34 percent.

The aft-diaphragm fastener clamp-up force will provide the necessary force to retain the airfoil shape for typical loading.

An arbitrary friction load of 35.72 kg/cm (200 lb/running in.) (which is almost twice maximum limit air load) was used for design of the leading edge and wing interface loads.

Figure 237 shows the typical leading-edge panel finite element model configuration, with the location of structural components and areas where margins were calculated.

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DRAWING	COMPONENT	MATERIAL ALLOY	MATERIAL FORM	REASON FOR SELECTION
LFC-19	MANIFOLDS			
	TUBE	2024-T3	TUBE 1.27/1.42 x 0.071cm (.5/.56 x .028 IN)	EASY TO FORM
	CONNECTOR	2024-T3	TUBE 0.635 x 0.071cm (.25 x .028 IN)	EASY TO FORM
	CONNECTOR	6061-T4	TUBE 0.635/0.787 x 0.071 cm (.25/.31 x .028 IN)	WELDABILITY
	RESTRICTOR	2024-T3	BAR 0.635cm (.25 IN DIA.)	EASY TO MACHINE
LFC-22	PICCOLO			
	TUBE	6061-T6	TUBE 1.113/1.27/1.42x0.071cm (.438/.5/.56 x .028 IN)	WELDABILITY
	FLEXIBLE TUBE			PER MIL-H-8794/MIL-H-5593

Figure 235. Tube Materials

DRAWING	COMPONENT	MATERIAL ALLOY	MATERIAL FORM	REASON FOR SELECTION
LFC-21	DIAPHRAGM & ANGLES			
	FORWARD			
	WEB	2024-TS	CLAD SHEET	LOW COST
	STIFFENER	7075-T6511	EXTRUSION	AVAILABLE
	LOWER ANGLE	301-1/4H	SHEET	FORMABILITY
	UPPER FITTING	GRAPHITE	FABRIC	FORMABILITY
	AFT			
	FACE SHEET	HMF 133/34	FABRIC	EASY TO FAB
	CORE	NOMEX-NYLON	10.16-0.318 x 1.57cm (4.0-1/8 x .62 IN)	COMPATIBILITY WITH G/E
	LOWER ANGLE	301-1/4H	SHEET	FORMABILITY

Figure 236. Materials for Diaphragms and Attaching Angles

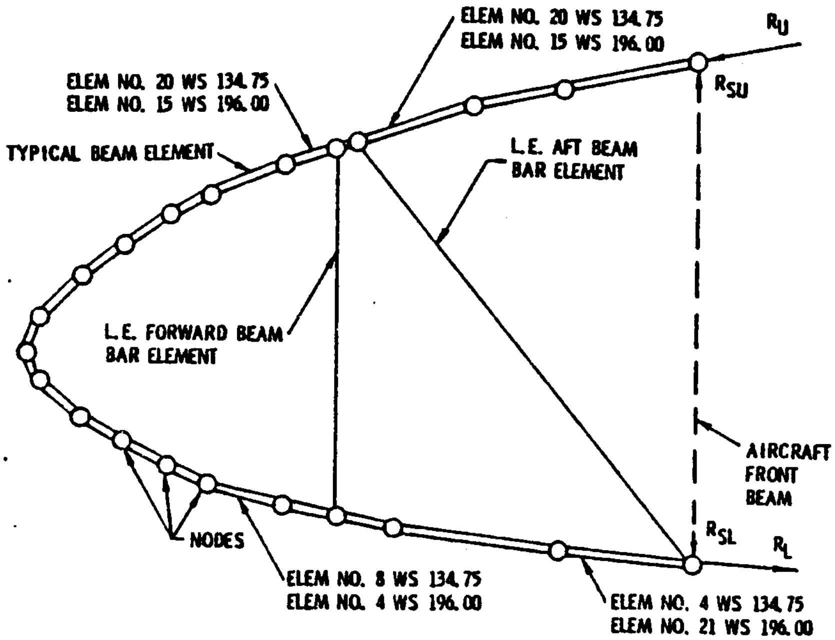

Figure 237. Finite Element Model

Five load cases were selected from the LFC JetStar load conditions, as shown in Table 25. Conditions CL-1 and CL-2 produce the highest suction and crushing pressures on each surface. Condition CL-3 was selected as the LFC baseline. Conditions CL-4 and CL-5 have downbending and upbending. Wing/leading-edge interface forced bending loads were combined with these loads.

TABLE 25. LOAD CONDITIONS

Case	Configuration	Mach	Angle of Attack	
			RAD	DEG
CL-1	Clean	0.2	0.314	18
CL-2	Clean	0.2	-0.249	-14
CL-3	LFC - 1 g Ve = 115.2 m/sec (224 KEAS)	0.75	0.071	+4.07
CL-4	Clean	0.80	0.070	-4
CL-5	Clean	0.85	0.122	+7

Figures 238 and 239 present a summary of critical internal limit loads obtained from the finite-element model. These loads are representative of the critical areas at four locations around the leading-edge section at each end of the test specimen. The "P" loads are tension (+) or compression (-) on the beam elements. The "V" loads are shear loads at each end of beam elements. The "M" loads are moments at each end of beam elements.

LOWER SURFACE

LOAD CONDITION	AFT			FWD		
	P	V	M	P	V	M
CL-1 +TEMP	3,993	2,294	-143	7,075	4,904	-233
CL-2 +TEMP	-1,366	2,119	127	-4,273	4,098	233
CL-3	-1,033	1,051	-29	-455	858	34
CL-4 +BEND +TEMP	25,236	4,151	484	-5,289	3,310	216
CL-5 +BEND +TEMP	-24,973	-3,170	-466	3,905	3,818	-187

LOADS (N/M)

MOMENTS (M-N/M)

UPPER SURFACE

LOAD CONDITION	AFT			FWD		
	P	V	M	P	V	M
CL-1 +TEMP	-6,953	2,802	320	1,243	-6,305	335
CL-2 +TEMP	8,319	3,240	-203	4,343	-3,485	-307
CL-3	-2,452	- 245	21	-1,296	88	24
CL-4 +BEND +TEMP	10,403	2,347	-167	3,608	-1,401	-303
CL-5 +BEND +TEMP	-11,646	-1,821	108	-4,536	2,399	317

LOADS (N/M)

MOMENTS (M-N/M)

Figure 238. Maximum Loads-WS134.75

LOWER SURFACE

LOAD CONDITION	AFT			FWD		
	P	V	M	P	V	M
CL-1 + TEMP	22.8	13.1	-32.2	40.4	28.0	-52.4
CL-2 + TEMP	-7.8	12.1	28.5	-24.4	23.4	52.4
CL-3	-5.9	6.0	-6.6	-2.6	4.9	-7.6
CL-4 + BEND + TEMP	144.1	23.7	108.8	-30.2	18.9	48.6
CL-5 + BEND + TEMP	-142.6	-18.1	-104.8	22.3	21.8	-42.1

LOADS (LB/IN)
MOMENTS (IN-LB/IN)

UPPER SURFACE

LOAD CONDITION	AFT			FWD		
	P	V	M	P	V	M
CL-1 + TEMP	-39.7	16.0	71.9	7.1	-36.0	75.4
CL-2 + TEMP	47.5	18.5	-45.6	24.8	-19.9	-69.1
CL-3	-14.0	-1.4	4.8	-7.4	0.5	5.4
CL-4 +BEND +TEMP	59.4	13.4	-37.5	20.6	-8.0	-68.1
CL-5 + BEND + TEMP	-66.5	-10.4	24.3	-25.9	13.7	71.2

LOADS (LB/IN)
MOMENTS (IN-LB/IN)

Figure 238. Maximum Loads-WS134.75 (Cont'd)

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LOWER SURFACE

LOAD CONDITION	AFT			FWD		
	P	V	M	P	V	M
CL-1 +TEMP	4,151	3,695	-199	6,375	-438	-195
CL-2 +TEMP	-2,469	718	101	-4,518	1,576	213
CL-3	-718	-53	-44	-263	-175	-29
CL-4 +BEND +TEMP	35,376	2,977	443	-6,077	1,629	214
CL-5 +BEND +TEMP	-36,286	-2,031	-500	4,501	-1,629	-181

UPPER SURFACE

LOAD CONDITION	AFT			FWD		
	P	V	M	P	V	M
CL-1 +TEMP	-11,646	1,839	150	2,644	-6,129	272
CL-2 +TEMP	11,804	2,522	-237	3,889	-736	-247
CL-3	-3,100	-261	22	-1,173	105	19
CL-4 +BEND +TEMP	16,585	1,681	-98	3,135	4,168	-268
CL-5 +BEND +TEMP	-17,933	-1,734	123	-4,693	1,016	219

LOADS (N/M)

MOMENTS (M-N/M)

Figure 239. Maximum Loads-WS196.00

LOAD CONDITION	LOWER SURFACE					
	AFT			FWD		
	P	V	M	P	V	M
CL-1 +TEMP	23.7	21.1	-44.8	36.4	-2.5	-43.8
CL-2 +TEMP	-14.1	4.1	22.6	-25.8	9.0	47.9
CL-3	-4.1	-.3	-9.8	-1.5	-1.0	-6.5
CL-4 +BEND +TEMP	202.	17.0	99.6	-34.7	9.3	48.2
CL-5 +BEND +TEMP	-207.2	-11.6	-112.4	25.7	-9.3	-40.6

LOAD CONDITION	UPPER SURFACE					
	AFT			FWD		
	P	V	M	P	V	M
CL-1 +TEMP	-66.5	10.5	33.8	-15.1	-35.	61.2
CL-2 +TEMP	67.4	14.4	-53.2	22.2	-4.2	-55.6
CL-3	-17.7	-1.49	5.0	-6.7	.6	4.3
CL-4 +BEND +TEMP	94.7	9.6	-22.1	17.9	23.8	-60.3
CL-5 +BEND +TEMP	-102.4	-9.9	27.7	-26.8	5.8	49.3

LOADS (LB/IN.)
MOMENTS (IN.-LB/IN.)

Figure 239. Maximum Loads-WS196.00 (Cont'd)

Figure 240 summarizes the leading-edge beam (diaphragm) limit loads corresponding to the five load conditions previously defined.

Figure 241 lists the leading-edge/wing-spar interface limit loads for each condition for each end of the test section.

Panel margins of safety are listed for four critical locations at each end of the LFC panel assembly, as shown in Figure 242. A factor-of-safety of 2.09 was included in the margins of safety. A "high" margin of safety is +1.00 or greater.

Figure 243 summarizes minimum margins of safety calculated in a detailed analysis of the LFC leading-edge panel installation.

8.3.4 Sonic Fatigue Analysis

The sonic fatigue analysis was based on predicted noise levels for the NASA JetStar and assumed an anticipated service life of 2700 hours at cruise noise levels, 100 hours at takeoff noise levels, and 10 hours at reverse-thrust noise levels. The highest noise level of 148 dB is obtained for the reverse-thrust case.

LOADS (N/M)

LOAD CONDITION	WS 134.75 FWD BEAM	WS 134.75 AFT BEAM	WS 196.00 FWD BEAM	WS 196.00 AFT BEAM
CL-1	-9,772	17,022	-9,404	17,127
CL-2	8,896	-15,289	8,249	-15,201
CL-3	-2,592	3,362	-2,697	3,520
CL-4	12,154	-58,667	15,551	-64,622
CL-5	13,029	58,667	-17,583	65,673

LOADS (LB/IN)

LOAD CONDITION	WS 134.75 FWD BEAM	WS 134.75 AFT BEAM	WS 196.00 FWD BEAM	WS 196.00 AFT BEAM
CL-1	-55.8	97.2	-53.7	97.8
CL-2	50.8	-87.3	47.1	-86.8
CL-3	-14.8	19.2	-15.4	20.1
CL-4	69.4	-335.	88.8	-369.
CL-5	74.4	335.	-100.4	375.

Figure 240. Diaphragm Loads

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LOADS (N/M)

LOAD CONDITION	WS 134.75				WS 196.00			
	R_U	R_{SU}	R_L	R_{SL}	R_U	R_{SU}	R_L	R_{SL}
CL-1	7,040	-1,296	11,226	13,905	11,576	753	14,903	12,504
CL-2	-8,441	1,366	-7,741	-12,452	-11,751	-753	-11,926	-10,928
CL-3	2,487	525	-18	3,047	3,170	1,068	1,226	3,082
CL-4	-26,970	613	-25,744	-43,782	-36,777	-228	-35,901	-42,731
CL-5	28,195	-560	23,642	44,657	38,178	403	34,500	44,832

LOADS - (LB/IN)

LOAD CONDITION	W.S. 134.75				W.S. 196.00			
	R_U	R_{SU}	R_L	R_{SL}	R_U	R_{SU}	R_L	R_{SL}
CL-1	40.2	-7.4	64.1	79.4	66.1	4.3	85.1	71.4
CL-2	-48.2	7.8	-44.2	-71.1	-67.1	-4.3	-68.1	-62.4
CL-3	14.2	3.0	-.1	17.4	18.1	6.1	7.0	17.6
CL-4	-154.	3.5	-147.	-250.	-210.	-1.3	-205.	-244.
CL-5	161.	-3.2	135.	255.	218.	2.3	197.	256.

Figure 241. Front Spar Loads

CRITICAL LOCATION	SAFETY MARGIN BEND + AXIAL	SAFETY MARGIN CORE SHEAR
W.S. 134.75 LWR AFT ELE #4	+.23	+.84
W.S. 134.75 LWR FWD ELE #8	+.91	+.57
W.S. 134.75 UPR AFT ELE #3	+HIGH	+HIGH
W.S. 134.75 UPR FWD ELE #20	+.61	+.22
W.S. 196.00 LWR AFT ELE #21	+.15	+HIGH
W.S. 196.00 LWR FWD ELE #4	+.92	+HIGH
W.S. 196.00 UPR AFT ELE #20	+HIGH	+HIGH
W.S. 196.00 UPR FWD ELE #15	+.75	+.26

Figure 242. Panel Margins of Safety

PART NAME	DRAWING NO.	MARGIN OF SAFETY
PANEL ASSEMBLY	LFC-18-1	.15
FORWARD DIAPHRAGM	LFC-21-11	.15
UPPER ATTACH ANGLE	LFC-21-29	.19
LOWER ANGLE	LFC-21-33	.09

Figure 243. Summary of Minimum Margins of Safety

Panel material allowables were obtained from the coupon test of SM-3, and the maximum stress levels for the panel, panel edge, and plumbing were calculated. Adequate positive margins of safety for the LFC leading-edge panel, panel edge, and plumbing were calculated.

9.0 DEVELOPMENT TESTS

The development tests described in this section where conducted to aid in the design and configuration selection of the internal plumbing and ducting of the leading-edge test article with respect to liquid, purge, and suction systems. All of the development tests were performed prior to the fabrication and assembly of the leading-edge flight test article.

Slot configuration (SC) tests were performed to aid in the development of a subsurface ducting configuration that would be relatively insensitive to spanwise pressure gradients and to evaluate candidate configurations. Tests were also conducted to evaluate cleaning/anti-icing fluids and fluid flow and coverage characteristics from the various candidate liquid slot designs. The tests conducted are coded as follows:

SC-1	Slot Flow Investigations
SC-2	Leading Edge Suction System Mock-Up Development
SC-3	Cleaning/Anti-Icing Liquid Properties
SC-4	Liquid Flow Characteristics
SC-5	Leading-Edge Cleaning System Mock-Up Development
-	Restrictor/Check Valve Selection

The primary purpose of the structures and materials (SM) series of tests was to verify the structural integrity of the LFC leading-edge panels. The structures and materials tests were:

SM-1	Adhesive Selection
SM-2	Test Panel Verification
SM-3	Coupon Flexural Fatigue Test
SM-5	Suction/Cleaning Line Fitting Test
SM-6	Full-Scale Static Ultimate Testing
SM-7	Sonic Fatigue Test
SC-3	Cleaning/Anti-Icing Liquid Properties (Lap Shear)

Two manufacturing development tests were conducted to develop procedures for slot cutting and measuring and to develop fabrication techniques. There tests were:

MD-1	Slot Cutting and Measuring Development
MD-2	Tooling and Manufacturing Development

9.1 SLOT FLOW INVESTIGATIONS (TEST SC-1)

Analytical predictions of the spanwise Cp variations for the original leading-edge glove configuration of the LEFT program were significantly larger than had been anticipated. These variations were reduced somewhat by airfoil redesign, but the variations still exceed levels expected for a production airplane. The original design for the slot/metering/ducting configuration was selected largely on the basis of relative ease of manufacture and insensitivity of slot spanwise suction flow distribution to the internal metering and ducting system. It became apparent that the predicted Cp variations for the LEFT airfoil would cause large spanwise variations in slot suction flow with the original slot/metering/ducting configuration. It was considered prudent to develop a configuration that would be more insensitive to external Cp variations. Test SC-1 specifically addressed the sensitivity of the spanwise distribution of slot suction flow to high spanwise variations in airfoil surface Cp.

9.1.1 Slot Suction Flow Sensitivity Test

The objective of test SC-1 was to explore sensitivity of slot suction flow to a pressure gradient on the slotted surface and to evaluate alternative configurations to desensitize suction flow to these pressure gradients with a minimum of modifications to the configuration. The test was to determine the spanwise slot suction flow distribution in the presence of a surface pressure gradient with various configurations of slot duct.

9.1.1.1 Slot Duct Configurations

The three slot duct configurations tested covered the range of duct cross-sectional areas being considered for the flight test article. A 25.4 cm x 40.6 cm (10 in x 16 in) flat test panel was fabricated of aluminum with the slot ducts, each 40.6 cm (16 in) long, machined into the top surface as shown in Figure 244.

The test ducts were sized as follows:

Slot Duct Configurations

Slot Duct	Width (cm)	Width (in)	Depth (cm)	Depth (in)	Area (cm^2)	Area (in^2)
A	0.381	0.150	0.203	0.080	0.088	0.01366
B	0.262	0.103	0.140	0.055	0.042	0.00645
C	0.142	0.056	0.076	0.030	0.012	0.00191

Slot duct A represents the original nominal configuration planned for the slot duct prior to the emergence of the significant spanwise Cp variations on the test surface. Slot duct B represents the duct design derived to minimize

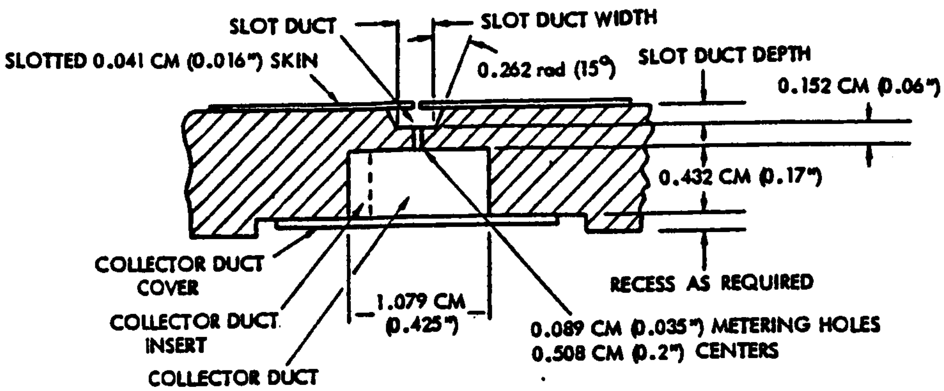

Figure 244. Basic Slot/Metering/Ducting Configuration for Test SC-1

the effects of the anticipated spanwise Cp variations at the time of the SC-1 tests. Slot duct C extends the configuration test range to even smaller slot duct cross sections. Corresponding collector ducts were machined into the reverse surface of the panel with the collector ducts recessed into the surface so as to provide the same material thickness between the slot duct and the collector duct. This was to provide for metering holes of a constant 0.152 cm (0.06 in) length for all slots. The collector duct widths were machined wider than the nominal 0.762 cm (0.30 in) width to provide for variation of collector duct size by inserting metal strips along the sides of the collector ducts to reduce the duct areas. The collector ducts had a nominal 0.432 cm (0.17 in) depth. Individual covers were made for each of the collector ducts and attached so they could be readily removed to alter the collector duct areas. The 0.089 cm (0.035 in) diameter metering holes were drilled at 0.508 cm (0.2 in) intervals along the centerline of each slot duct. The panel was covered with 0.041 cm (.016 in) aluminum sheet bonded to the panel surface and slots cut along the slot duct centerlines with nominally 0.010 cm (0.004 in.) saws.

9.1.1.2 Surface Pressure Variation

The surface pressure gradient along a slot was adjusted incrementally by means of a pressure distribution box. The box had 5 compartments of 7.62 cm (3 in) length and each compartment was provided with a static pressure tap and an inlet port. The inlet ports were connected through control valves to a flow-meter. The test set-up with the pressure distribution box in position over a slot is shown in Figure 245.

More localized flow distributions were recorded with the distribution box removed (no pressure gradient) by using a probe made of tygon tubing connected to a rotameter.

Suction was applied to the appropriate collector duct of the slot being tested at a single point near the end of the collector duct. Some additional testing was conducted by applying suction to the collector duct at two points spaced to simulate the actual configuration of the test article.

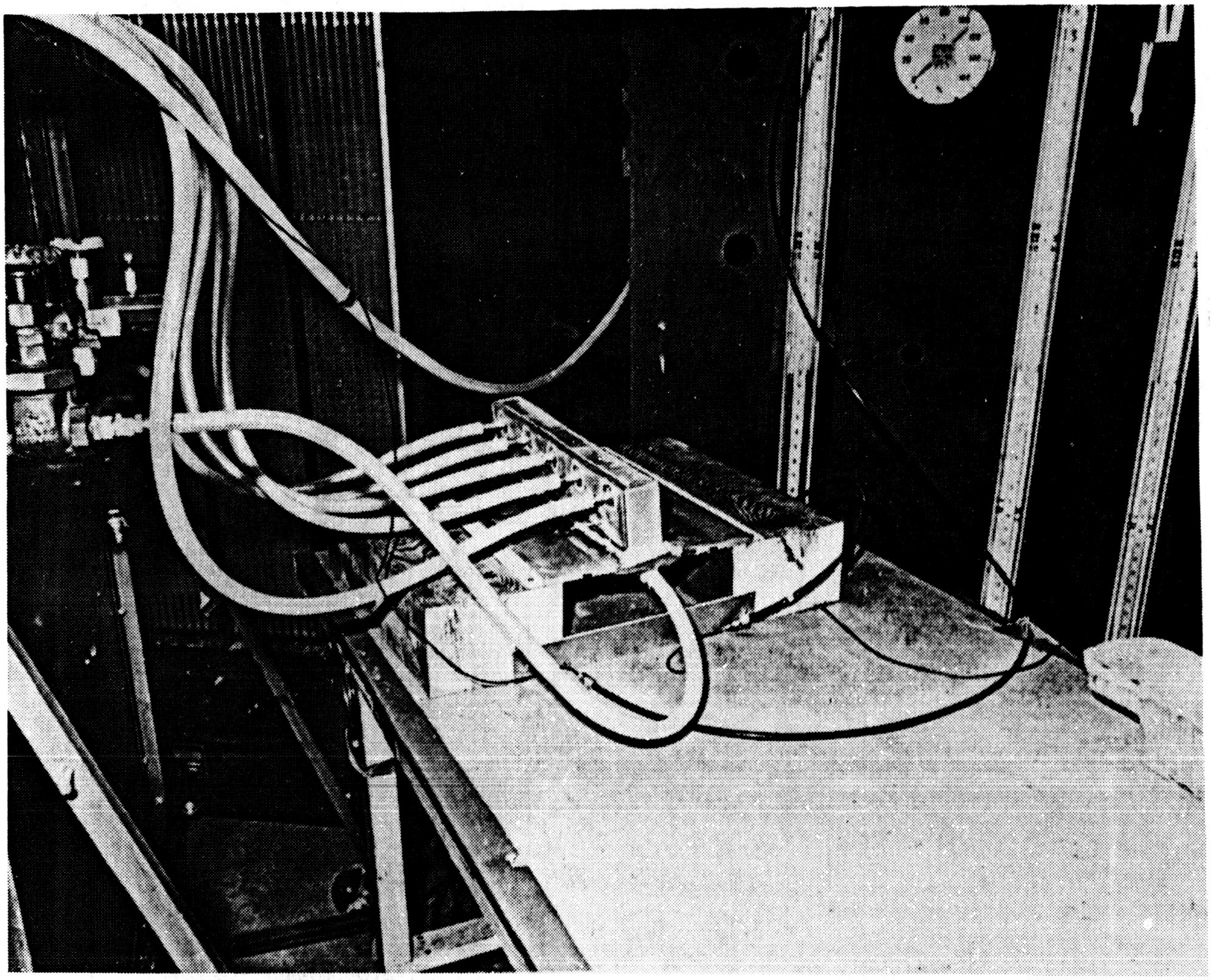

Figure 245. Pressure Distribution Box

9.1.2 SC-1 Test Results

The results of the SC-1 test demonstrate the effects of a spanwise wing-surface-pressure gradient (Cp variation) on slot flow variation and indicate how slot ducts and collector ducts may be sized to reduce the sensitivity of spanwise slot flow to spanwise pressure gradients. Similarly, these data indicate the degree of control over slot spanwise flow variation that may be obtained by sizing slot and collector ducts for a particular pressure distribution, flow level, and collector duct outlet spacing. Figure 246 illustrates typical results obtained with suction applied to one end of the collector duct. Suction was applied at the end of the test panel having the lowest surface pressure, where surface pressure was controlled by the compartmented pressure distribution box. The flow into each of the five compartments of the distribtuion box is plotted on the vertical axis of all figures shown. This flow was controlled to maintain a constant differential pressure between adjacent compartments. The pressure differentials used were 2.54, 5.08 and 7.62 cm (1, 2, and 3 in) of water, which correspond approximately to spanwise wing-surface-pressure co-

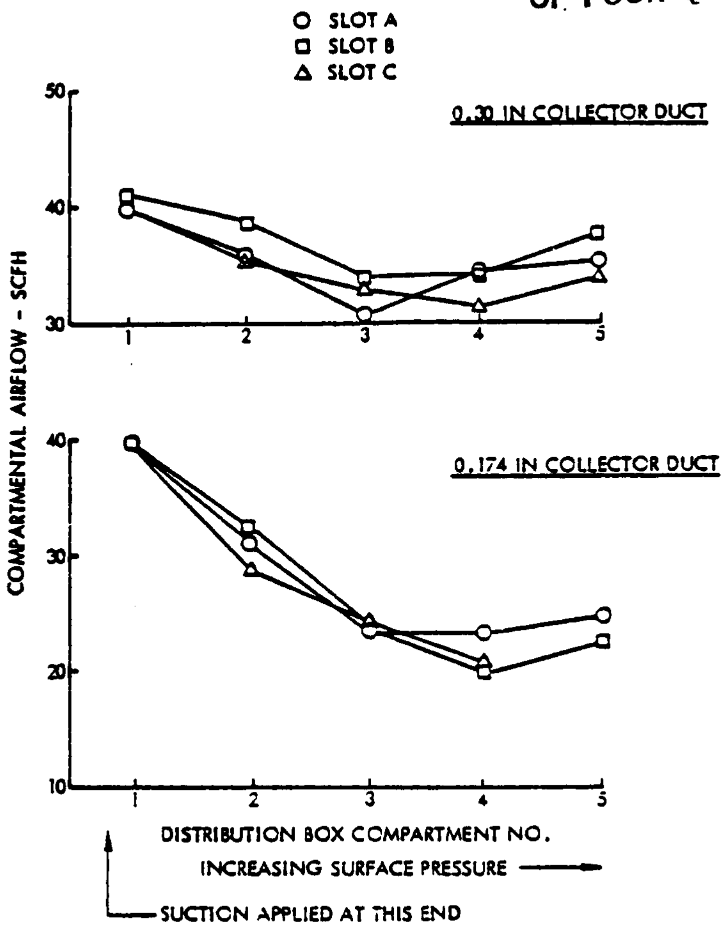

Figure 246. Effect of Slot Duct Size on Slot Flow Distribution

efficient gradients of 0.006, 0.014 and 0.020 per meter (0.02, 0.045, and 0.065 per foot) of span, respectively. Slot duct B, with a collector duct width of 0.762 cm (0.30 in), was closest to the then current design configuration. The surface pressure gradient tends to produce increasing flow in the spanwise direction from Box 1 to Box 5. The peak local flow may occur at either end of the panel depending on the particular duct configuration and pressure gradient combination.

Figure 246 indicates the effect of slot duct size on flow distribution. The slot duct cross-sectional areas for slots A, B and C are 0.088, 0.0416 and 0.0123 cm^2 (0.01366, 0.00645, and 0.00191 in^2), respectively. It was found that slot duct size has much less effect on flow distribution than does collector duct size. The range of configurations tested with the localized suction appear to have little spanwise flow in the slot duct, and consequently little sensitivity to slot duct size.

As indicated previously, some additional testing was accomplished on slot ducts A and B with suction applied to the collector duct at two locations. These suction locations were selected to simulate the planned spacing of 15.24 cm (6 in). The two locations reduce flow path lengths within the collector duct, and thus reduce the influence of collector duct size and reduce the spanwise flow variation.

For the configuration most representative of the then current design [slot B with collector duct width of 0.762 cm (0.30 in)], suction at two locations reduces the flow variation by about one-half at a nominal wing Cp variation of about 0.0137 per meter (0.045 per foot) of span. The effect of slot duct size appears to remain small.

9.1.2.1 Selected Slot Duct Geometry

It was concluded that the configuration of slot B with the collector duct width of 0.762 cm (0.30 in), represented a good compromise for a slot/metering geometry between insensitivity to spanwise Cp variations and acceptable slot flow stability characteristics. With respect to the latter, a semi-circular slot duct geometry of 0.152 cm (0.060 in) radius was selected. This shape produces a duct spanwise flow area about 12 percent smaller than Slot B while increasing the slot duct height by 9 percent. The decision to select this shape was also influenced by the geometry of the test article substructure. By previous design, the concave recess designed to accept the slot duct favored a slot duct insert of semi-circular shape. Also of major significance, Lockheed-funded experiments in slot flow stability being conducted concurrently with test SC-1 indicated the slot velocity stability characteristics of the 0.152 cm (0.060 in) radius slot duct were acceptable. This assessment is applicable for slot Reynolds number up to and including those anticipated for the LEFT design point 50 percent oversuction condition.

9.2 LEADING-EDGE SUCTION SYSTEM MOCK-UP DEVELOPMENT (TEST SC-2)

In the early stages of the test article design, candidate configurations were developed for suction slots, cleaning/anti-icing slots, and dual-purpose slots. All configurations are required to collect and transport suction airflow to the slot flow control valves inside the JetStar fuselage. In addition, several configurations are required to deliver the cleaning/anti-icing liquid to the test surface. The various configurations evolved from analytical studies, tests of some, but not all, components of the complete system, and the assembly/space requirements of the test articles. Furthermore, in some cases, test measurements of component pressure losses were found to differ from analytical predictions by an amount that may be of little consequence for some applications but may be significant for the LFC test article. None of the candidate configurations had previously been tested as a complete system for slot flow distribution or flow/pressure loss characteristic with fluid flow in the cleaning/anti-icing mode (where appropriate), or with air flow in the suction or purge mode. Thus, it was highly desirable that the complete candidate systems be flow tested prior to commitment to the final designs. Test SC-2 was designed to accomplish this task with respect to air flow in the suction and purge modes. The purge requirements and the purge capability of the candidate configurations are explored in test SC-5, described later. Thus, the primary objectives of the SC-2 test were to (1) verify the analytical procedures used to predict system component pressure losses, and (2) evaluate local slot flow variations.

9.2.1 Test Article Description (SC-2)

The full-scale wing-section model built for the Phase I low-speed wind-tunnel suction and washing tests as described in NASA CR 159253, September 1980, was used for this test. This test specimen consists of a two-dimensional air-

foil leading-edge model having an active suction area of about 121.92 cm (48 in) spanwise and 38.1 cm (15 in) chordwise. The model was modified for testing of each of the candidate configuration options for collecting suction flow from dedicated suction slots, cleaning/anti-icing slots, and dual-purpose slots. The existing surface skin was removed and selected slots were modified to the latest slot duct design, i.e. semi-circular duct of 0.152 cm (0.060 in) radius. The surface was re-skinned and slots cut to the current design slot width of 0.0094 cm (0.0037 in.). The model was also modified to simulate the slot lines running from the test article to the slot flow control valve inside the test vehicle fuselage. These lines included bulkhead fittings at the inboard end of the test piece and at the pressure vessel interface. The test set-up is illustrated in Figure 247.

Local slot flow distribution for each slot/metering/ducting configuration was measured during selected suction and purge runs by the same flowmeter/probe device used during the SC-1 test. During suction or purge operation, the probe was placed over the slot on the model surface at several locations along the slot span to determine slot flow distribution.

The following five slot configurations were included in the test:

Slot Indent.	Type of Manifold*	Configuration
U1	Corrugated Duct	Suction only
U3	Piccolo tube	Suction only
P2	Piccolo tube	Dual-purpose using restrictor/check valves
W2	Corrugated Duct	Cleaning/anti-icing only
W4	Piccolo tube	Cleaning/anti-icing only

*Connection between slot collector duct and suction system at test article interface.

Comparisons between measured and predicted pressure losses were obtained for the following ducting system components:

(1) Line from test panel interface to flowmeter including bulkhead fitting (Simulated slot lines Numbers 1 and 2)

(2) Manifold tube

(3) Flow passage between manifold tube and collector duct (piccolo flex air-hose or metering hole)

(4) Slot plus metering orifices.

These system components are shown schematically in Figure 247.

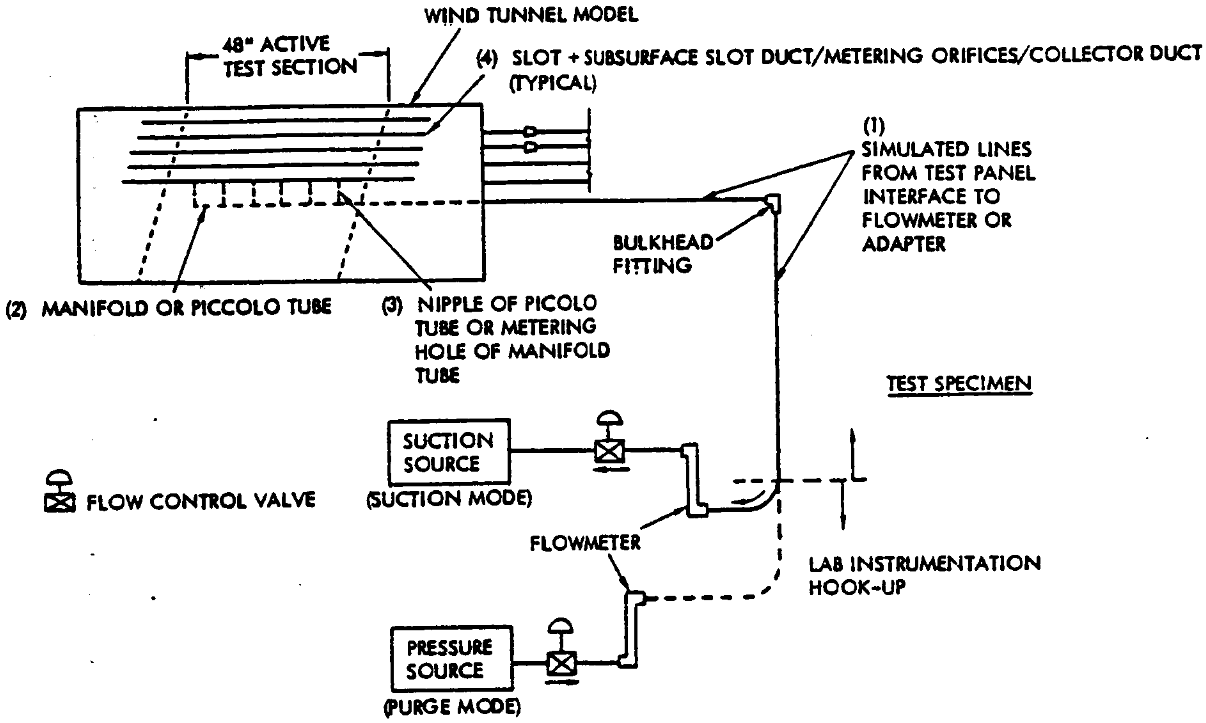

Figure 247. SC-2 Test Set-Up Schamatic

9.2.2 SC-2 Test Results

The results of the pressure loss comparisons among the different slot configurations in both suction and purge modes were very similar. In some cases, the analytical procedures produced results that were in close agreement with the measured data. In other cases, similar comparisons showed less than satisfactory agreement. In all cases where significant differences in analytical and experimental results were found, the appropriate computer code was modified to provide better agreement with the measured data. These computer codes, modified when necessary, were subsequently used in developing the suction line sizes for both the Lockheed and McDonnell Douglas systems.

The level of agreement obtained between the measured and predicted results for the four system segments listed previously varied from good to unacceptable. The line losses showed good agreement and the analytical procedure employed to predict line losses was retained without modification. The measured pressure losses in the manifold tubes of the configurations tested indicated the computer code underestimated this loss. A previous comparison using Lockheed in-house experimental data indicated good agreement with predicted levels for the losses in a complete and similar piccolo tube configuration. Nonetheless, the analytical procedures were modified to generate manifold tube losses more in agreement with the results of SC-2. The predicted losses for the piccolo tube nipples and the metering holes of manifold tubes bonded directly to the leading edge substructure showed reasonably good agreement with the measured data. The piccolo tube nipples of slot R2 include the check valve losses. Slight modifications were made to the computer codes to improve these comparisons.

The most notable discrepancy in the comparisons of predicted and measured data occurred in the slot plus metering orifice pressure losses. The results indicated measured losses of about 7 percent compared to 1.5 percent for predicted losses at a typical design flow rate. This result was not expected. Previous analyses of available in-house data for slots and metering holes had indicated the prediction equations were conservative relative to measured data. Furthermore, results from a suction test on an LFC test panel conducted as a part of the LFC Wing Panel Program, a separate NASA contract, show good agreement for these same comparisons. The conclusion was that the measured slot plus metering orifice data for the SC-2 test were in error and that the prediction procedure is reliable.

The slot local flow investigations during SC-2 indicated acceptable results. Several of the test configurations demonstrated rather significant local flow variation, but this was attributed, in part, to fairly large variations in slot widths. In some cases, visual inspection alone confirmed sizeable slot width variations. It was also suspected that some metering orifice blockage may have occurred from excess adhesive during model fabrication.

9.3 CLEANING/ANTI-ICING LIQUID PROPERTIES (TEST SC-3)

The cleaning/anti-icing fluid used during the Phase I wind tunnel test described in NASA Contractor Report 159253, September 1980, exhibited unacceptable physical characteristics. The liquid failed to evaporate from the surface and slots following a washing sequence and its freezing point was considered too high to provide adequate icing protection. This series of tests was directed toward selecting a suitable liquid for the cleaning and anti-icing functions based on physical/chemical properties and characterisitcs. These properties include flash point, fire point, viscosity, volatility, freezing point, surface tension, thermal stability, residue characteristics, corrosive or reactivity effects on aircraft or LFC subsystems materials, and toxic properties.

9.3.1 Cleaning Fluid Selection

From the initial group of candidate fluids, which included a PGME liquid solution submitted by McDonnell Douglas, one was selected for further investigation. This was a fluid based on ethylene glycol monobutyl ether (EGBE). Other fluids were tentatively rejected on the basis of a low flash point or residue levels. For example, the surfectants in the initial test group were found to leave unacceptable residues, while the ethers were found to have relatively low flash points. Subsequent testing of the EGBE fluid showed that water does not form an azetrope with EGBE. This ruled out the EGBE/water solution as an anti-icing fluid. Other glycol ethers do not exhibit this characteristic. A restudy of possible fluids was initiated. This new study again included PGME although this fluid has a lower flash point and lower viscosity than desired. This fluid had been selected by McDonnell Douglas for use in their contamination avoidance liquid system.

Subsequent testing did not reveal a completely satisfactory cleaning/anti-icing/de-icing fluid. None of the glycol ethers appeared satisfactory as a de-icing fluid but some appeared marginally acceptable as an anti-icing fluid. The decision was made to use a 60/40 solution of PGME and water (McDonnell

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Douglas selected fluid) which appeared to be the best compromise available at the time. There were also obvious advantages to using the same liquid in both systems. Having selected a fluid, exposure testing was begun using lap shear specimens representing the graphite/epoxy fabric structure to titanium skin bond. The computed stress for all test specimens exceeded the design allowable of 6,205 KPa (900 psi) with a high margin of safety. Testing by McDonnell Douglas had also shown a good resistance of epoxy materials to PGME. Thus, no degradation of the FM123-4 bond is anticipated with exposure to the 60/40 PGME solution, and this fluid has been selected as the cleaning/anti-icing liquid for the LEFT program.

9.4 LIQUID FLOW CHARACTERISTICS (TEST SC-4)

The purpose of this test was to establish liquid flow characteristics and capabilities for candidate cleaning/anti-icing liquids with the latest slot configuration. The cleaning liquid mixtures used during these tests were selected from the mixtures of SC-3, see Section 9.3, judged at the time to be most compatile with cleaning/anti-icing system requirements. The cleaning liquids selected were ethylene-glycol/water (E-G) and propylene-glycol-methyl-ether/water (PGME) liquid mixtures. The E-G mixture was mixed with 50 percent water, and 0.5 percent tergitol as a wetting agent. A very small portion of sodium fluorescent dye was added to each mixture to enhance the photographs used to record liquid coverage of the test model surface.

9.4.1 SC-4 Test Article Description

The model used for this test series was the cylindrical leading edge model used in the Phase I tests as reported in NASA Contractor Report 159253, September 1980, with a new slot/metering orifice test specimen. The model configuration mounted in the test tunnel is shown in Figure 248. The leading-edge surface has a sweep angle of 0.48 rad (27.5 deg) and a variable

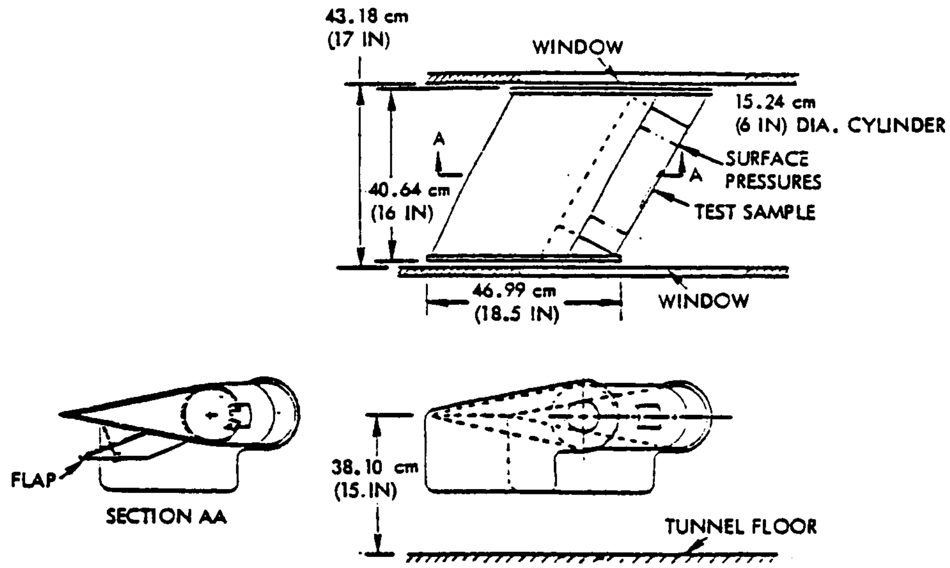


Figure 248. Cleaning Liquid Test Model

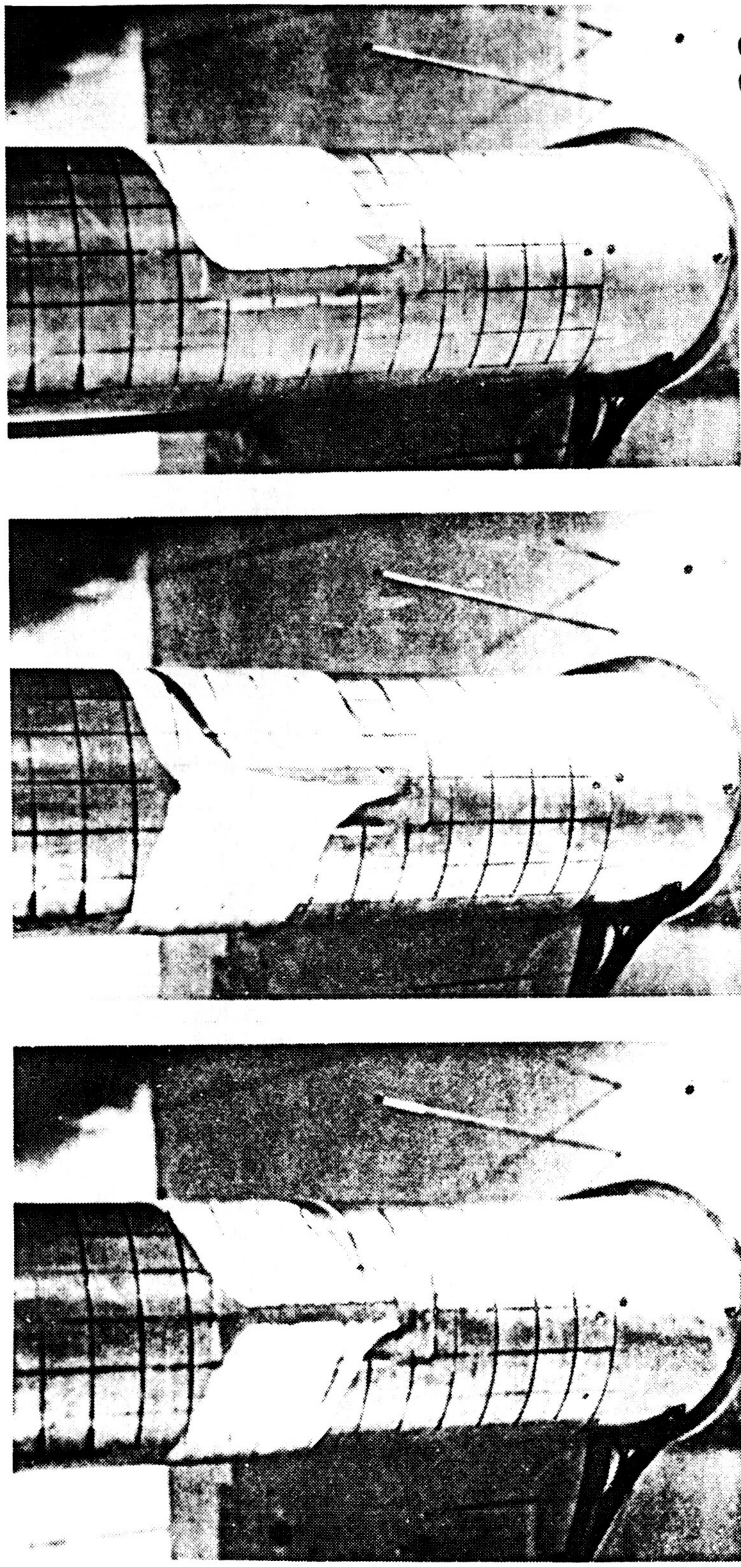
stagnation point with respect to slot location. The 15.24 cm (6 in) diameter cylinder forms the leading edge with flat-plate aft fairings tangent to the cylindrical surface. A recess is machined into the cylinder to form a chamber for injecting liquid on to the surface. The front of this recess is provided with a recessed flange such that a 12.7 cm (5 in) by 5.08 cm (2 in) circumferential surface test specimen can be flush-bonded over the recess.

The test specimen had dual 0.010 cm (0.004 in) slots centered at a 1.57 cm (0.62 in) spacing. A pressurized liquid injection chamber is provided inside the cylinder, which permits flow through a perforated plate into a small plenum behind the metering orifice channel. Design configuration orifices of 0.076 cm (0.030 in) diameter set at a 0.51 cm (0.2 in) pitch were used to meter flow into the channel under each skin slot. The test specimen skin being recessed in the cylinder surface was positioned by cylinder rotation to locate the airflow stagnation point at the desired test location with respect to the slots. The model was instrumented with static pressure taps in circumferential rows near both ends of the test segment to establish a given test condition Cp distribution and stagnation point location. A 2.54 cm (1 in) grid was painted on the cylinder surface to aid in evaluation of cleaning liquid coverage.

The model was mounted in a small wind tunnel having side-wall windows for photographic coverage of the test section. An ultra-violet lamp was used to hi-lite the liquid flow pattern (fluorescent dye) in the photographs. The wind tunnel has a 60.96 cm x 60.96 cm (24 in x 24 in) test section. Calibrated tunnel velocities up to a tunnel dynamic pressure (q) range of 0.316 KPa (6.6 psf) were available. The model has a movable trailing-edge flap such that airflow around the model can be forced to flow at varied angles, thus permitting movement of the stagnation point with respect to the model centerline. A flowmeter was used to determine the liquid flow rate at each test condition. The bulk of these tests covered varied flow rates and stagnation point locations with the PGME liquid mixture. This PGME mixture is the same as the McDonnell Douglas system liquid, -- 60 percent PGME, 40 percent water. Results typical of these tests are shown in Figure 249. The flow patterns shown are for the design liquid slot flow rate of 0.0277 Kg/min/cm (0.024 lb/min/in) and a tunnel dynamic pressure (q) of 0.316KPa (6.6 psf).

9.4.2 SC-4 Test Results

The observed results indicated that PGME has a better pattern spread and provides a more consistent coverage through wider variations of stagnation point. The most forward slot pair will have a dry region initially but upon lift-off and rotation there will be coverage of all regions between slots through the climb period due to split flows at one slot or another. Leading edge rotation which moves the stagnation point in a range of approximately $\pm$0.051 cm ($\pm$0.02 in) above or below a slot will cause slot flow splits to occur from one or more of the forward slots. Photos were obtained for a sequence of flow conditions to illustrate the flow patterns through a range of stagnation point locations relative to the two forward-most slots. This sequence covered the two extremes of both slots having downward flow (stagnation just above upper slot) to both slots having upward flow (stagnation point just below bottom slot). Figure 249 illustrates the latter case. It was also observed that the PGME dried faster after shut-down and did not appear to leave a surface residue.



m_{liq} = 0.024 lb/min/in

q_{tun} = 6.0 psi

Figure 249. PGME Liquid Flow Tests-Stagnation Down

It was concluded from the results of test SC-4 that the PGME liquid exhibited acceptable liquid flow characteristics with respect to the requirements of the LEFT test article.

9.5 LEADING-EDGE CLEANING SYSTEM MOCK-UP DEVELOPMENT (TEST SC-5)

The primary objective of this test was to evaluate liquid flow distribution during static simulation of airplane take-off and climb-out attitudes and accelerations. System pressure loss vs flow rate and purge requirements were also to be evaluated.

9.5.1 Test Article Description

The test hardware consisted of the Phase I wind-tunnel model described in Section 9.2. The test fluid was a 60/40 percent mixture of PGME and water. Aircraft takeoff acceleration and climb-out attitude were simulated by inclining the lateral axis of the leading-edge model to simulate internal liquid pressure heads.

9.5.2 SC-5 Test Results

Photographs were used to evaluate liquid flow distribution on the model surface. A small amount of fluorescent dye was added to the test fluid to enhance the photos. The configuration tested was that of the dual-purpose slots with restrictor/check valves. The photos were evaluated qualitatively for the effects of flow rate and inclination angle. Flow rate was judged to have no significant effect on flow distribution. Figure 250, for example, shows generally good flow distribution at a low flow, i.e. flow is about evenly distributed spanwise along the slot even though the fluid only covers a small percentage of the surface. In the presence of freestream flow, based on the results of Phase I tests and test SC-4, the fluid rivulets seen in the photo would tend to coalesce and complete coverage of the surface would be expected.

Inclination angle had only a small effect on flow distribution. Even though flow distribution was judged satisfactory at all angles, some improvement was noted at increasing angles. At zero inclination, the inboard half of the slot received somewhat more flow than the outboard half. Inclining the wing shifted the flow pattern to the low (outboard) end to a more even distribution.

The results of the flow and pressure loss measurements on SC-5 proved inconclusive. It appears that several instrumentation lines may have contained trapped air and thus biased the liquid pressure readings. Though inconsistent, the data did indicate a potential problem with check valve operation in the resistor (liquid flow) mode. During some runs, it was apparent that rapid increases in flow caused most if not all check valves to close (the desired response) while a gradual rise in flow to the same level closed few, if any, check valves. This suggests special attention to check valve design/selection and installation to avoid this potential problem. The result of improper operation as described above on the flight test article would be an excessive cleaning fluid usage rate but otherwise the system would be expected to perform satisfactorily. Furthermore, during testing, the model was oriented in the most unfavorable position for check valve operation. It was positioned so the liquid flow direction through the check valves was vertically upward. In the actual

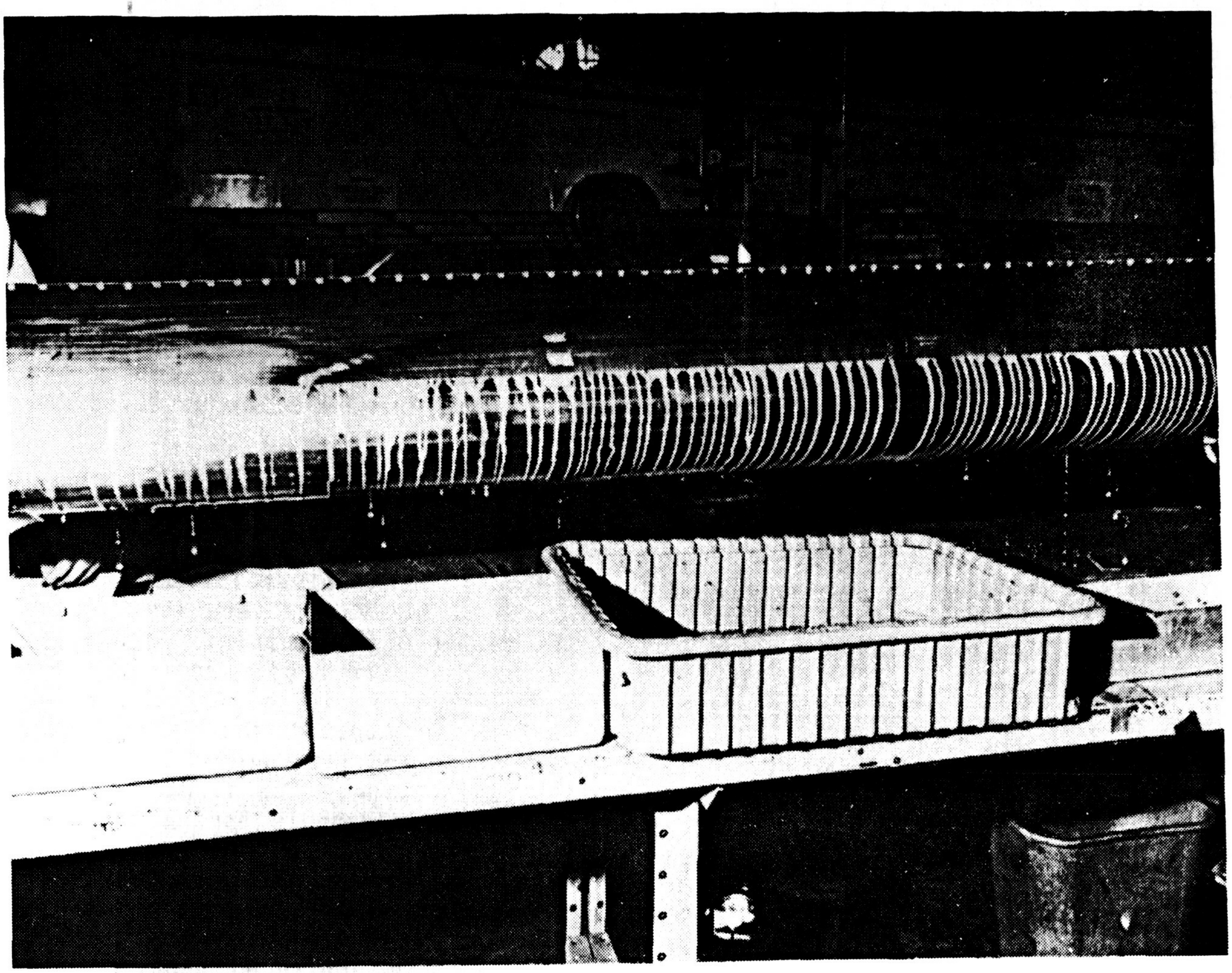


Figure 250. Typical Photo for Flow Distribution Evaluation

installation the check valves are positioned no closer to vertical than about 1.05 rad (60 deg) with respect to the wing reference plane. In the actual positions, gravity will assist in holding the check valve poppets at the proper position for liquid system operation. As noted previously, no problem with adequate flow distributed was indicated.

The results of the purge evaluation indicated marginal purge capability for the dual-purpose slot and adequate capability for a suction-only slot. The following result is typical of the vent and purge operation observed on the dual-purpose slot. The wing inclination angle was 0.07 rad (4 deg). The system lines were filled with liquid (PGME) and then vented at 13.79 KPa (2 psi) differential for about 3 to 4 minutes. The vent operation applied suction to the system in an attempt to pull residual fluid back into the liquid supply tank. This operation removed approximately 80 percent of the fluid that could be removed by venting. At the end of the vent operation, the slot appeared on the average about 75 percent clear of fluid. Venting was followed by purge air flow at 20.68 KPa (3 psig) for 10 minutes. A small amount of additional liquid was removed and the slot appeared about 40 percent clear of fluid. Increasing purge to 41.37 KPa (6 psig) for 8 minutes removed another small amount of fluid and the slot appeared about 80 percent clear with some fluid still dripping from

the low end of the slot. Purging without a vent sequence cleared the system except for some residual dripping at the slot in approximately 97 seconds. This occurred at 13.79 KPa (2 psi) with check valves apparently open and at 20.68 KPa (3 psi) with check valves apparently closed. The results were somewhat inconsistent, but it appears that the vent operation saves fluid but does not reduce purge time. It should be noted, since this test was run, the check valve restrictor orifice size has been increased from about 0.10 cm (0.040 in) to about 0.11 cm (0.044 in) diameter. This size increase is expected to improve the purge capability of the dual-purpose slots.

Purge capability for suction-only slots with limited fluid contamination was also evaluated and found to be adequate. Liquid was poured over one of the slots so that only the slot was filled with fluid. Purge airflow at 20.68 KPa (3 psig) for 5 minutes completely cleared the slot.

9.6 RESTRICTOR/CHECK VALVE SELECTION

A test evaluation was made of three candidate check valves for use in the combination suction/cleaning slots. The poppets of all valves were modified by drilling a 0.10 cm (0.040 in) hole through the center of the poppet. Initially, the springs in each valve were not removed so the valves were spring-loaded closed during cleaning flow operation with the cleaning liquid metered by the 0.10 cm (0.040 in) center hole. When suction is applied the valves open permitting relatively unrestricted suction flow. The end fittings of all valves were machined for clamp-on attachment of flexible tubing. The first of these valves had a 20.68 KPa (3 psi) opening spring with adjustable spring tension. This valve was acceptable but had the highest pressure loss characteristic even with the spring adjusted for zero preload and was subsequently rejected for this reason. The second valve had a 2.30 KPa (0.333 psi) opening pressure spring and was larger than the other two valves. Although this valve had the largest internal passages and the lowest pressure losses, it chattered resulting in unacceptable suction flow pulses. The third valve also had 2.30 KPa (0.333 psi) opening pressure spring, was smaller than the preceding valve, and exhibited higher pressure losses. This valve also showed unacceptable chattering characteristics throughout all but the very highest range of desired suction flows. Removing the spring from this valve cured the chattering and also reduced the pressure differential required to crack the valve open with application of suction. The characteristics of this valve with liquid flow were satisfactory and were unchanaged whether the spring was present or not, even when oriented in a position requiring the liquid to lift the poppet in order to close it. Since space is at a premium near the nose of the test article, this valve, Nupro Model B-4CP2-1, without a spring, was selected for inclusion in the SC-2 and SC-5 tests. This check valve proved acceptable throughout the subsequent testing and the Nupro A-4CP2-1, without a spring, was specified for the flight test article. This is the same valve but is made of aluminum rather than brass.

9.7 ADHESIVE SELECTION (TEST SM-1)

The primary purpose of these tests was to evaluate FM123-4 and AF163-2 adhesives which were candidates to bond the external titanium skin to the hybrid laminate leading-edge surface panels. After slotting the wind tunnel leading-edge test article of Phase I some slot closures resulted. Slot closure could be

a result of residual stress between the titanium skin and the graphite panel during hot bond due to the difference in thermal expansion of these materials.

One way to reduce these thermal stresses is to lower the cure temperature from 93.3°C (200°F) to 82.2°C (180°F). No design data are available at these lower cure temperatures. These tests developed design data by testing fifty lap shear specimens in a tensile-shear mode.

Five specimens of each of the two adhesive systems were evaluated under the following environmental conditions:

DRY:	Cold = -53.9°C (-65°F)	WET:	Cold = -53.9°C (-65°F)
	Room temperature		Hot = 71.1°C (160°F)
	Hot = 71.1°C (160°F)		

9.7.1 SM-1 Summary of Results

In the SM-1 adhesive selection tests, all of the specimens using all the FM123-4 film adhesive cured at 82.2°C (180°F) met or exceeded the design allowable shear stress of 6,205 KPa (900 psi), except for three of the wet-hot specimens. The worst-case failure occurred at a shear stress of 1,379 KPa (200 psi). It was concluded that FM123-4 adhesive cured at 82.2°C (180°F) does not meet the lap shear requirements, as follows:

SM-1 - Adhesive Selection

- FM123-4 Cured at 82.2°C (180°F)
  Failed - Low Shear Stress

- FM123-4 Cured at 93.3°C (200°F)
  Acceptable Shear Stress

The AF163-2, a high flow adhesive, was used as a duplex system on the LFC Phase I test article in which a good fillet bead was required. This characteristic is not required for the current design. In the Wing Surface Structural Development (WSSD) project tests using FM123-4 adhesive cured at 93.3°C (200°F), the lowest lap shear failure occurred at 8,273 KPa (1200 psi), which exceeds requirements for the LEFT project. Therefore, this adhesive is used with a curing temperature of 93.3°C (200°F).

9.7.2 Detail Results of SM-1 Adhesive Selection

Fifty test specimens, using FM123-4 and AF163-2 film adhesive cured at 82.2°C (180°F), were fabricated. Specimen geometries are presented in Tables 26 and 27.

Test results for the FM123-4 adhesive specimens listing failure loads, modes of failure, and the computed failure stresses are presented in Table 28. The design allowable shear stress is 6,205KPa (900 psi). All specimens met or exceeded the 6,205 KPa (900 psi) level except for three of the wet-hot specimens. The worst case failure occurring at a shear stress of 1,379 KPa (200 psi). These failures occurred in the adhesive and/or the adhesive/titanium

TABLE 26. SPECIMEN-FM123-4 ADHESIVE (82°C/180°F)

SPECIMEN NUMBER	JOINT THICKNESS t_1 - CM	GRAPHITE THICKNESS t_2 - CM	TITANIUM THICKNESS t_3 - CM	BONDLINE THICKNESS CM	JOINT LAP LENGTH L - CM	JOINT LAP WIDTH W - CM	JOINT LAP AREA CM^2
F-1 DR	0.429	0.300	0.114	0.015	1.321	2.540	3.355
F-2 DR	0.427	0.300	0.109	0.018	1.295	2.548	3.300
F-3 DR	0.427	0.302	0.109	0.015	1.295	2.550	3.302
F-4 DR	0.409	0.297	0.109	0.003	1.245	2.548	3.172
F-5 DR	0.419	0.300	0.112	0.008	1.346	2.548	3.430
F-1 DC	0.434	0.302	0.112	0.020	1.321	2.548	3.366
F-2 DC	0.345	0.221	0.114	0.010	1.346	2.545	3.426
F-3 DC	0.424	0.302	0.114	0.008	1.270	2.548	3.236
F-4 DC	0.419	0.300	0.109	0.010	1.270	2.548	3.236
F-5 DC	0.424	0.302	0.109	0.018	1.295	2.548	3.300
F-1 DH	0.345	0.224	0.114	0.008	1.321	2.548	3.366
F-2 DH	0.345	0.224	0.114	0.008	1.346	2.548	3.430
F-3 DH	0.424	0.302	0.112	0.010	1.321	2.550	3.369
F-4 DH	0.422	0.300	0.112	0.010	1.295	2.548	3.300
F-5 DH	0.434	0.305	0.114	0.015	1.321	2.548	3.366
F-1 WC	0.345	0.226	0.114	0.005	1.372	2.545	3.492
F-2 WC	0.348	0.226	0.114	0.008	1.372	2.545	3.492
F-3 WC	0.417	0.302	0.109	0.005	1.270	2.548	3.236
F-4 WC	0.434	0.305	0.112	0.018	1.270	2.548	3.236
F-5 WC	0.417	0.302	0.112	0.003	1.295	2.548	3.300
F-1 WH	0.345	0.224	0.114	0.008	1.321	2.548	3.366
F-2 WH	0.345	0.224	0.114	0.008	1.346	2.548	3.430
F-3 WH	0.424	0.302	0.112	0.010	1.321	2.550	3.369
F-4 WH	0.422	0.300	0.112	0.010	1.295	2.548	3.300
F-5 WH	0.434	0.305	0.114	0.015	1.321	2.548	3.366

SPECIMEN NUMBER	JOINT THICKNESS t_1 (IN)	GRAPHITE THICKNESS t_2 (IN)	TITANIUM THICKNESS t_3 (IN)	BONDLINE THICKNESS (IN)	JOINT LAP LENGTH L (IN)	JOINT LAP WIDTH W (IN)	LAP JOINT AREA (IN^2)
F-1 DR	.169	.118	.045	.006	0.52	1.000	0.52
F-2 DR	.168	.118	.043	.007	0.51	1.003	0.51
F-3 DR	.168	.119	.043	.006	0.51	1.004	0.51
F-4 DR	.161	.117	.043	.001	0.49	1.003	0.49
F-5 DR	.165	.118	.044	.003	0.53	1.003	0.53
F-1 DC	.171	.119	.044	.008	0.52	1.003	0.52
F-2 DC	.136	.087	.045	.004	0.53	1.002	0.53
F-3 DC	.167	.119	.045	.003	.050	1.003	0.50
F-4 DC	.165	.118	.043	.004	0.50	1.003	0.50
F-5 DC	.167	.119	.043	.007	0.51	1.003	0.51
F-1 DH	.136	.088	.045	.003	0.52	1.003	0.52
F-2 DH	.136	.088	.045	.003	0.53	1.003	0.53
F-3 DH	.167	.119	.044	.004	0.52	1.004	0.52
F-4 DH	.166	.118	.044	.004	0.51	1.003	0.51
F-5 DH	.171	.120	.045	.006	.052	1.003	0.52
F-1 WC	.136	.089	.045	.002	0.54	1.002	0.54
F-2 WC	.137	.089	.045	.003	0.54	1.002	0.54
F-3 WC	.164	.119	.043	.002	0.50	1.003	0.50
F-4 WC	.171	.120	.044	.007	0.50	1.003	0.50
F-5 WC	.164	.119	.044	.001	0.51	1.003	0.51
F-1 WH	.136	.088	.045	.003	0.52	1.003	0.52
F-2 WH	.136	.088	.045	.003	0.53	1.003	0.53
F-3 WH	.167	.119	.044	.004	0.52	1.004	0.52
F-4 WH	.166	.118	.044	.004	0.51	1.003	0.51
F-5 WH	.171	.120	.045	.006	0.52	1.003	0.52

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TABLE 27. SPECIMEN-AF163-2 ADHESIVE (82°C/180°F)

SPECIMEN NUMBER	JOINT THICKNESS t_1 - CM	GRAPHITE THICKNESS t_2 - CM	TITANIUM THICKNESS t_3 - CM	BONDLINE THICKNESS CM	JOINT LAP LENGTH L - CM	JOINT LAP WIDTH W - CM	JOINT LAP AREA CM^2
A-1 DR	0.348	0.224	0.112	0.013	1.524	2.545	3.876
A-2 DR	0.356	0.226	0.112	0.018	1.473	2.548	3.753
A-3 DR	0.356	0.221	0.112	0.023	1.422	2.545	3.619
A-4 DR	0.353	0.224	0.112	0.008	1.372	2.545	3.492
A-5 DR	0.340	0.226	0.112	0.008	1.372	2.548	3.496
A-1 DC	0.340	0.224	0.112	0.005	1.346	2.543	3.423
A-2 DC	0.340	0.224	0.112	0.005	1.346	2.548	3.430
A-3 DC	0.351	0.224	0.112	0.016	1.346	2.548	3.430
A-4 DC	0.351	0.226	0.112	0.003	1.346	2.545	3.426
A-5 DC	0.351	0.224	0.112	0.005	1.346	2.545	3.426
A-1 DH	0.345	0.221	0.114	0.010	1.372	2.545	3.492
A-2 DH	0.353	0.224	0.114	0.005	1.346	2.545	3.426
A-3 DH	*	0.221	0.112	*	1.448	2.543	3.682
A-4 DH	*	0.224	0.112	*	1.448	2.545	3.685
A-5 DH	0.353	0.221	0.109	0.013	1.346	2.548	3.430
A-1 WC	0.353	0.226	0.112	0.005	1.346	2.545	3.426
A-2 WC	0.340	0.224	0.112	0.005	1.346	2.545	3.426
A-3 WC	*	0.226	0.112	*	1.448	2.545	3.685
A-4 WC	*	0.224	0.112	*	1.397	2.543	3.553
A-5 WC	0.353	0.226	0.112	0.005	1.372	2.548	3.496
A-1 WH	0.338	0.224	0.109	0.005	1.295	2.545	3.296
A-2 WH	0.353	0.224	0.112	0.008	1.397	2.543	3.553
A-3 WH	*	0.226	0.112	*	1.346	2.543	3.423
A-4 WH	*	0.221	0.112	*	1.346	2.543	3.423
A-5 WH	0.353	0.226	0.109	0.008	1.346	2.540	3.419

*UNABLE TO MEASURE DUE TO ADHESIVE ON TITANIUM

TABLE 27. SPECIMEN-AF163-2 ADHESIVE (82°C/180°F) (CONT'D)

SPECIMEN NUMBER	JOINT THICKNESS t_1 (IN)	GRAPHITE THICKNESS t_2 (IN)	TITANIUM THICKNESS t_3 (IN)	BONDLINE THICKNESS (IN)	JOINT LAP LENGTH L (IN)	JOINT LAP WIDTH W (IN)	JOINT LAP AREA (IN^2)
A-1 DR	.137	.088	.044	.005	0.60	1.002	0.60
A-2 DR	.140	.089	.044	.007	0.58	1.003	0.58
A-3 DR	.140	.087	.044	.009	0.56	1.002	0.56
A-4 DR	.135	.088	.044	.003	0.54	1.002	0.54
A-5 DR	.134	.089	.044	.003	0.54	1.003	0.54
A-1 DC	.134	.088	.044	.002	0.53	1.001	0.53
A-2 DC	.134	.088	.044	.002	0.53	1.003	0.53
A-3 DC	.138	.088	.044	.006	0.53	1.003	0.53
A-4 DC	.138	.089	.044	.001	0.53	1.002	0.53
A-5 DC	.138	.088	.044	.002	0.53	1.002	0.53
A-1 DH	.136	.087	.045	.004	0.54	1.002	0.54
A-2 DH	.135	.088	.045	.002	0.53	1.002	0.53
A-3 DH	*	.087	.044	*	0.57	1.001	0.57
A-4 DH	*	.088	.044	*	0.57	1.002	0.57
A-5 DH	.135	.087	.043	.005	0.53	1.003	0.53
A-1 WC	.135	.089	.044	.002	0.53	1.002	0.53
A-2 WC	.134	.088	.044	.002	0.53	1.002	0.53
A-3 WC	*	.089	.044	*	0.57	1.002	0.57
A-4 WC	*	.088	.044	*	0.55	1.001	0.55
A-5 WC	.135	.089	.044	.002	0.54	1.003	0.54
A-1 WH	.133	.088	.043	.002	0.51	1.002	0.51
A-2 WH	.135	.088	.044	.003	0.55	1.001	0.55
A-3 WH	*	.089	.044	*	0.53	1.001	0.53
A-4 WH	*	.087	.044	*	0.53	1.001	0.52
A-5 WH	.135	.089	.043	.003	0.53	1.000	0.53

*UNABLE TO MEASURE DUE TO ADHESIVE ON TITANIUM

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TABLE 28. SPECIMEN TEST RESULTS-FM123-4 ADHESIVE (82°C/180°F)

SPECIMEN NUMBER	MAXIMUM LOAD (KG)	MAXIMUM STRESS (KPa) X 10^{-3}	FAILURE MODE - % ADHESIVE TO GRAPHITE	ADHESIVE TO TITANIUM	COHESIVE	GRAPHITE
F-1 DR	528.4	15.17				100
F-2 DR	694.0	20.68				100
F-3 DR	621.4	18.62				100
F-4 DR	755.2	23.44				100
F-5 DR	759.8	22.06				100
F-1 DC	526.2	15.17		95	5	
F-2 DC	616.9	17.93	30	70		
F-3 DC	367.4	11.03		100		
F-4 DC	419.6	12.41		100		
F-5 DC	421.8	12.41		100		
F-1 DH	653.2	19.30			100	
F-2 DH	635.0	17.93			100	
F-3 DH	449.0	13.10			100	
F-4 DH	478.5	14.48			100	
F-5 DH	476.3	13.79			100	
F-1 WC	410.5	11.72	20	80		
F-2 WC	494.4	13.79	20	80		
F-3 WC	272.2	8.27	10	90		
F-4 WC	503.5	15.17		80	20	
F-5 WC	215.5	6.21	20	80		
F-1 WH	308.4	8.96	100			
F-2 WH	45.4	1.38		100		
F-3 WH	97.5	2.76		100		
F-4 WH	326.6	8.96	100			
F-5 WH	99.8	2.76		90	10	

TABLE 28. SPECIMEN TEST RESULTS-FM123-4 ADHESIVE (82°C/180°F) (CONT'D)

SPECIMEN NUMBER	MAXIMUM LOAD (LB)	MAXIMUM STRESS (KSI)	FAILURE MODE - % ADHESIVE TO GRAPHITE	ADHESIVE TO TITANIUM	COHESIVE	GRAPHITE
F-1 DR	1165	2.2				100
F-2 DR	1530	3.0				100
F-3 DR	1370	2.7				100
F-4 DR	1665	3.4				100
F-5 DR	1675	3.2				100
F-1 DC	1160	2.2		95	5	
F-2 DC	1360	2.6	30	70		
F-3 DC	810	1.6		100		
F-4 DC	925	1.8		100		
F-5 DC	930	1.8		100		
F-1 DH	1440	2.8			100	
F-2 DH	1400	2.6			100	
F-3 DH	990	1.9			100	
F-4 DH	1055	2.1			100	
F-5 DH	1050	2.0			100	
F-1 WC	905	1.7	20	80		
F-2 WC	1090	2.0	20	80		
F-3 WC	600	1.2	10	90		
F-4 WC	1110	2.2		80	20	
F-5 WC	475	0.9	20	80		
F-1 WH	680	1.3	100			
F-2 WH	100	0.2		100		
F-3 WH	215	0.4		100		
F-4 WH	720	1.3	100			
F-5 WH	220	0.4		90	10	

bond. It is concluded that FM123-4 adhesive cured at 82.2°C (180°F) does not meet the lap shear requirements.

Test results for the AF163-2 adhesive specimens are shown in Tables 29 and 30. AF163-2, a high-flow adhesive, was used as a duplex system on the LFC Phase I full-scale wind-tunnel model, in which a good fillet bead was required. The current slot duct has a 0.15 cm (0.06 in) radius semi-circle that does not require a good fillet bead. Thus, the AF163-2 adhesive was not required in LEFT project fabrication.

TABLE 29. SPECIMEN TEST RESULTS-AF163-2 ADHESIVE (82°C/180°F)

SPECIMEN NUMBER	MAXIMUM LOAD (KG)	MAXIMUM STRESS (KPa) X 10^{-3}	FAILURE MODE - % ADHESIVE TO GRAPHITE	FAILURE MODE - % ADHESIVE TO TITANIUM	FAILURE MODE - % COHESIVE	FAILURE MODE - % GRAPHITE
A-1 DR	938.9	23.44	20			80
A-2 DR	911.7	24.13	50			50
A-3 DR	809.7	22.06	10			90
A-4 DR	979.8	27.58	30			70
A-5 DR	952.5	26.89	85			15
A-1 DC	612.3	17.24		100		
A-2 DC	635.0	17.93	5	95		
A-3 DC	573.8	16.55	20	80		
A-4 DC	601.0	17.24		100		
A-5 DC	657.7	18.62		100		
A-1 DH	718.9	19.99			100	
A-2 DH	709.9	19.99			100	
A-3 DH	764.3	20.68			100	
A-4 DH	784.7	20.68			100	
A-5 DH	737.1	21.37			100	
A-1 WC	188.2	5.52	5	95		
A-2 WC	238.1	6.89		100		
A-3 WC	358.3	10.34	30	70		
A-4 WC	371.9	10.34	20	80		
A-5 WC	551.1	15.17	100			
A-1 WH	342.5	10.34	80	20		
A-2 WH	356.1	9.65	80	20		
A-3 WH	358.3	10.34	70	30		
A-4 WH	337.9	9.65	80	20		
A-5 WH	464.9	13.10	95	5		

TABLE 29. SPECIMEN TEST RESULTS-AF163-2 ADHESIVE (82°C/180°F) (CONT'D)

SPECIMEN NUMBER	MAXIMUM LOAD (LB)	MAXIMUM STRESS (KSI)	FAILURE MODE - %			
			ADHESIVE		COHESIVE	GRAPHITE
			TO GRAPHITE	TO TITANIUM		
A-1 DR	2070	3.4	20			80
A-2 DR	2010	3.5	50			50
A-3 DR	1785	3.2	10			90
A-4 DR	2160	4.0	30			70
A-5 DR	2100	3.9	85			15
A-1 DC	1350	2.5		100		
A-2 DC	1400	2.6	5	95		
A-3 DC	1265	2.4	20	80		
A-4 DC	1325	2.5		100		
A-5 DC	1450	2.7		100		
A-1 DH	1585	2.9			100	
A-2 DH	1565	2.9			100	
A-3 DH	1685	3.0			100	
A-4 DH	1730	3.0			100	
A-5 DH	1625	3.1			100	
A-1 WC	415	0.8	5	95		
A-2 WC	525	1.0		100		
A-3 WC	790	1.5	30	70		
A-4 WC	820	1.5	20	80		
A-5 WC	1215	2.2	100			
A-1 WH	755	1.5	80	20		
A-2 WH	785	1.4	80	20		
A-3 WH	790	1.5	70	30		
A-4 WH	745	1.4	80	20		
A-5 WH	1025	1.9	95	5		

In fabricating the test articles, FM123-4 adhesive cured at 93.3°C (200°F) is used. This selection is based on WSSD project tests using FM123-4 adhesive cured at 93.3°C (200°F) for bonding titanium to graphite tape. These test results are shown in Table 31. The lowest lap shear failure occurred at 8,273 KPa (1200 psi), which exceeds requirements for the LEFT project.

9.8 TEST PANEL VERIFICATION (TEST SM-2)

Development test SM-2 was conducted to verify the structural integrity of the leading-edge panel. Tests required for this verification are outlined as follows:

TABLE 30. SUMMARY TEST RESULTS-FM123-4 AND AF163-2 ADHESIVES (82°C/180°F)

SPECIMEN NUMBER	AVERAGE STRESS KPa X 10^{-3} (KSI)	FAILURE MODEL
F1-5 DR	19.99 (2.9)	100% GRAPHITE
F1-5 DC	13.79 (2.0	93% ADHESIVE TO TITANIUM, 6% ADHESIVE TO GRAPHITE, 1% COHESIVE
F1-5 DH	15.86 (2.3)	100% COHESIVE
F1-5 WC	11.03 (1.6)	82% ADHESIVE TO TITANIUM, 14% ADHESIVE TO GRAPHITE, 5% COHESIVE
F1-5 WH	4.83 (0.7)	58% ADHESIVE TO TITANIUM, 40% ADHESIVE TO GRAPHITE, 2% COHESIVE
A1-5 DR	24.82 (3.6)	61% GRAPHITE, 39% ADHESIVE TO GRAPHITE
A1-5 DC	14.24 (2.5)	95% ADHESIVE TO TITANIUM, 5% ADHESIVE TO GRAPHITE
A1-5 DH	20.68 (3.0)	100% COHESIVE
A1-5 WC	9.65 (1.4)	69% ADHESIVE TO TITANIUM, 39% ADHESIVE TO GRAPHITE
A1-5 WH	10.39 (1.5)	81% ADHESIVE TO GRAPHITE, 19% ADHESIVE TO TITANIUM

(1) Spanwise bending, shear, and compression

(2) Chordwise bending, shear, and compression

(3) Flatwise tension

9.8.1 SM-2 Summary of Results

Three groups of panel verification specimens were tested, including column compression, four point bending, and flatwise tension. Table 32 presents these results.

TABLE 31. SUMMARY TEST RESULTS-FM123-4 AT 93.3°C (200°F) CURE

SPECIMEN NUMBER	BONDLINE THICKNESS (CM)	LAP AREA (CM^2)	MAXIMUM LOAD (KG)	MAXIMUM STRESS (KPa X 10^{-3})	FAILURE MODE - % ADHESIVE TO TITANIUM	COHESIVE
MV2-1 FMWH	0.005	2.903	331.1	11.03	20	80
MV2-2 FMWH	0.010	3.419	294.8	8.27	30	70
MV2-3 FMWH	0.013	3.548	408.2	11.03	50	50
MV2-4 FMWH	0.013	3.419	419.6	11.72	10	90
MV2-5 FMWH	0.010	3.419	344.7	9.65	50	50
AVERAGE				10.34	32	68

TABLE 31. SUMMARY TEST RESULTS-FM123-4 AT 93.3°C (200°F) CURE (CONT'D)

SPECIMEN NUMBER	BONDLINE THICKNESS (IN)	LAP AREA (IN^2)	MAXIMUM LOAD (LB)	MAXIMUM STRESS (KSI)	FAILURE MODE (PERCENT) ADHESIVE TO TITANIUM	FAILURE MODE (PERCENT) COHESIVE
MV2-1 FMWH	.002	0.45	730	1.6	20	80
MV2-2 FMWH	.004	0.53	650	1.2	30	70
MV2-3 FMWH	.005	0.55	900	1.6	50	50
MV2-4 FMWH	.005	0.53	925	1.7	10	90
MV2-5 FMWH	.004	0.53	760	1.4	50	50
AVERAGE				1.5	32	68

TABLE 32. SM-2 TEST RESULTS

	Spanwise	Failing Load Kg	Failing Load (Lb)	Chord-wise	Failing Load * Kg	Failing Load * (Lb)
Column Compression	SC-1	8,618	19,000	CC-1	2,563	5,650
	SC-2	9,684	21,350	CC-2	2,926	6,450
	SC-3	10,546	23,250	CC-2	3,925	7,550

* Failed within 1 percent of predicted failure load

	Spanwise	Kg	(Lb)	Chord-wise	Kg	(Lb)
Four Point Bending	SB-1	1,297	2,860	CB-1	349	770
	SB-2	1,161	2,560	CB-2	321	707 **
	SB-3	1,197	2,640	CB-3	352	775

** 2.1 times design allowable

		Failing Stress KPa	Failing Stress (lb/in^2)
Flatwise Tension	FT-1/2	2,068	300*
	FT-3/4	2,758	400

* Equal to design allowable

9.9 COUPON FLEXURAL FATIGUE TEST (TEST SM-3)

The objectives of the SM-3 coupon flexural fatigue test are outlined below:

(1) Determine the fundamental cantilever resonant frequency

(2) Determine the deflection and strain properties at resonance

(3) Determine the strain-level/fatigue-life curve

9.9.1 SM-3 Summary of Results

The SM-3 specimen size is 4.83 x 26.67 x 1.27 cm (1.9 x 10.5 x 0.5 in). Test specimens were cantilevered on both ends between clamp blocks which were attached to an electromagnetic vibrator. The first two specimens were used as trial tests to establish the strain levels for the other group of specimens.

A gradual and continuous accumulation of internal damage was evident. From the start of each test, resonance frequency would gradually deteriorate. Also, corresponding increases in damping were observed early in the tests. However, visible damage was not evident until the advanced stages of fatigue. Generally, the tests were continued until there was visible evidence of damage or the resonance frequency had decreased about 10 percent. Both usually occurred about the same time. Catastrophic failure never occurred. In all cases, at termination of the testing, the specimens could have withstood additional strain cycles at the high load conditions. Invariably, the titanium sheet appears to separate from the graphite beneath it, usually at the slot closest to the root of the cantilever beam.

A curve of strain level vs. cycles to failure was established using a 5 percent reduction of resonance frequency (10 percent reduction in stiffness) as the failure criteria. At the point where resonance frequency had decreased 5 percent, a substantial increase in damping was generally apparent, indicating significant internal damage, even if external changes were not obvious.

The strain-life curve so obtained is shown in Figure 251. As expected, the data scatter is greater than usually seen for homogeneous materials. Considering the complexity of the leading-edge structure, it is nominal. The inverse negative slope, α, of the ϵ-N curve is computed to be 13.6, which is typical for graphite-epoxy skin laminates.

These results have been used for preliminary checks of the leading-edge sonic fatigue life, and were used to properly relate the leading-edge sonic fatigue test life to the design service life. This is necessary because the sonic fatigue test was an accelerated-life test.

9.10 SUCTION/CLEANING LINE TO PANEL FITTING (TEST SM-5)

During the fabrication and testing of the Phase I leading-edge test article, several problems were encountered with the plastic lines and fittings used for both suction and cleaning system plumbing. Examples of these problems include; the breakage of the line-to-panel fitting joints, resulting in leaks, and the crimping of plastic lines, resulting in restricted fluid and suction flow. To avoid such problems in the current test article design, rigid lines

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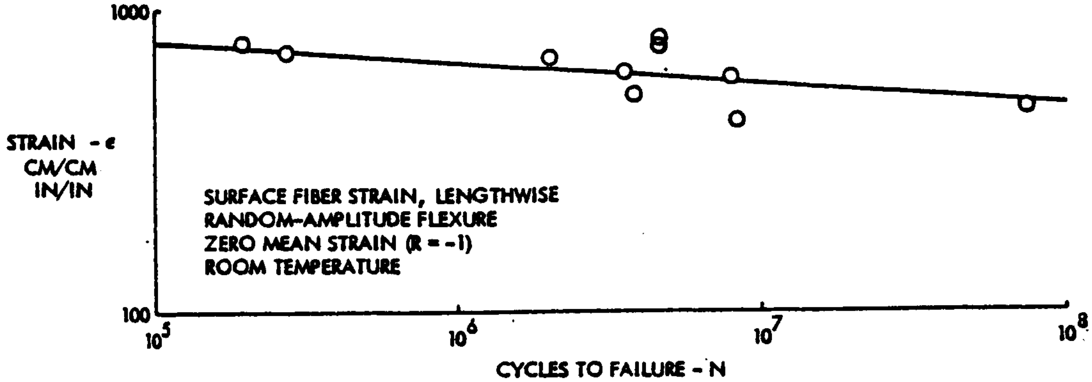


Figure 251. Fatigue Life of LFC Leading-Edge Composite Structure

and improved fitting joints are used. The objective of this test was to verify the structural integrity of the current suction/cleaning line fitting design. A component specimen which represents the suction/cleaning line to the leading-edge panel fitting was fabricated and tested, see Figure 252.

9.10.1 SM-5 Summary of Results

Eight test specimens were fabricated; four specimens were tested in tension and four were tested in bending.

Test results for the tension tests are summarized in Table 33.

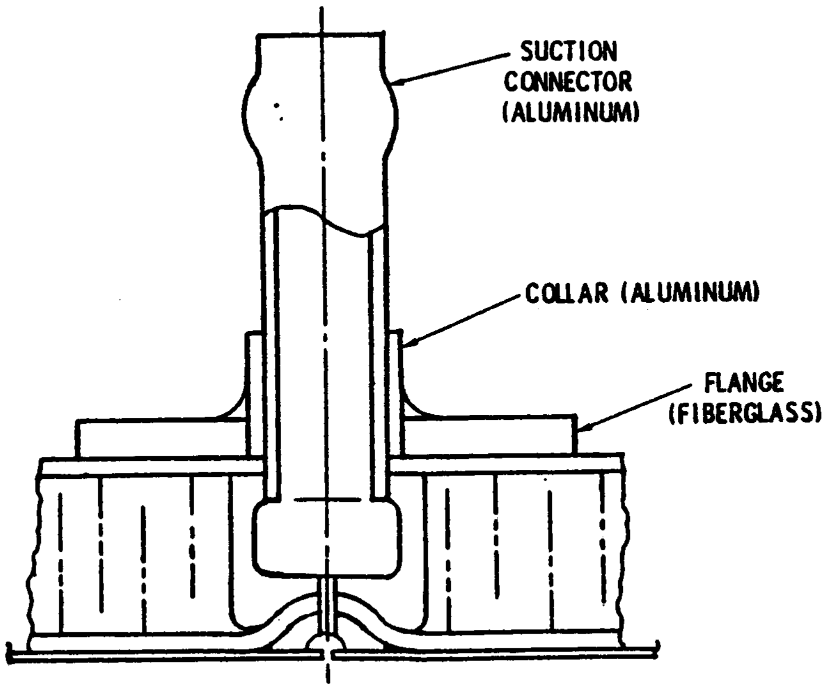


Figure 252. Suction/Cleaning Line to Panel Fitting

C-4

TABLE 33. SM-5 TEST RESULTS (TENSION)

Specimen Number	Failing Tension Load Kg	(lb)	Failure Mode
1	209	460	Tube to fiberglass bond
2	188	414	Tube to fiberglass bond
3	240	530	100 percent adhesion to graphite failure
4	207	456	Load jig failure

The design goal for the failing tension load was established at 22.7 Kg (50 lb). Therefore, the suction/cleaning line fitting far exceeds the design tension requirements.

Test results for the bending tests are summarized in Table 34.

TABLE 34. SM-5 TEST RESULTS (BENDING)

Specimen Number	Maximum Load Kg	(lb)	Comments and Failure Mode
1	20	44	fitting broke at saw cut*
2	24	52	fitting bent at saw cut*
3	23	50	fitting bent at saw cut*
4	15	32	fitting broke at saw cut*

*a 0.795 cm OD x 7.62 cm long (0.313 in OD x 3.0 in long) aluminum tube was placed over and bonded to the test tube fitting 0.635 x 1.905 cm long (10.25 in tube x 0.75 in long) in order to apply the required shear load at 3.81 cm (1.5 in) from the panel. The test tube was butted against the 0.795 cm OD x 3.81 cm long (0.313 in OD x 0.250 in long) collar. It was decided to saw cut the loading tube back away from the collars to make the test more representative; however, resulting saw cut nicks may have affected the test results.

Specimen numbers 2 and 3 withstood the design requirements of 22.7 Kg (50 lb) on a 3.81 cm (1.5 in) arm, or a bending moment equal to 8.47 N-M (75 in-lb); therefore, LFC leading-edge suction/cleaning line fittings successfully met the bending requirements.

9.11 FULL-SCALE STATIC ULTIMATE TESTING (TEST SM-6)

The leading-edge section, length = 66.5 cm (26.2 in), was loaded to simulate upper and lower surface pressures greater than the maximums expected during flight test. The chordwise pressure distributions applied are shown in Figures 253 and 254. The distributions were uniform in the spanwise direction except for a 2.54 cm (1 in) gap in the center to provide room for dial gauges. No damage was detected during or after loading.

9.11.1 Test Article Description and Test Procedure

The leading-edge section was attached to an aluminum plate which was bolted to the vertical wall of the universal test frame. Attachment was such that the leading-edge section was in an inverted position, thus allowing the lower surface to be loaded with lead shot and lead pigs. Pads were bonded to the upper surface, and loads were applied by two hydraulic actuators through linkages designed to provide the desired distribution. Also, part of the upper surface loading was applied by lead shot in the forward, center, and aft bays of the test panel. A total shot weight of 182.1 Kg (401.5 lb) was uniformly distributed spanwise in the bays. Figures 255 and 256 show the loading arrangement in more detail.

Three dial indicators were mounted at center span to measure leading edge deflections. The exact indicator locations are shown in Figure 257. Also shown in Figure 257 are locations for 15 axial strain gauges.

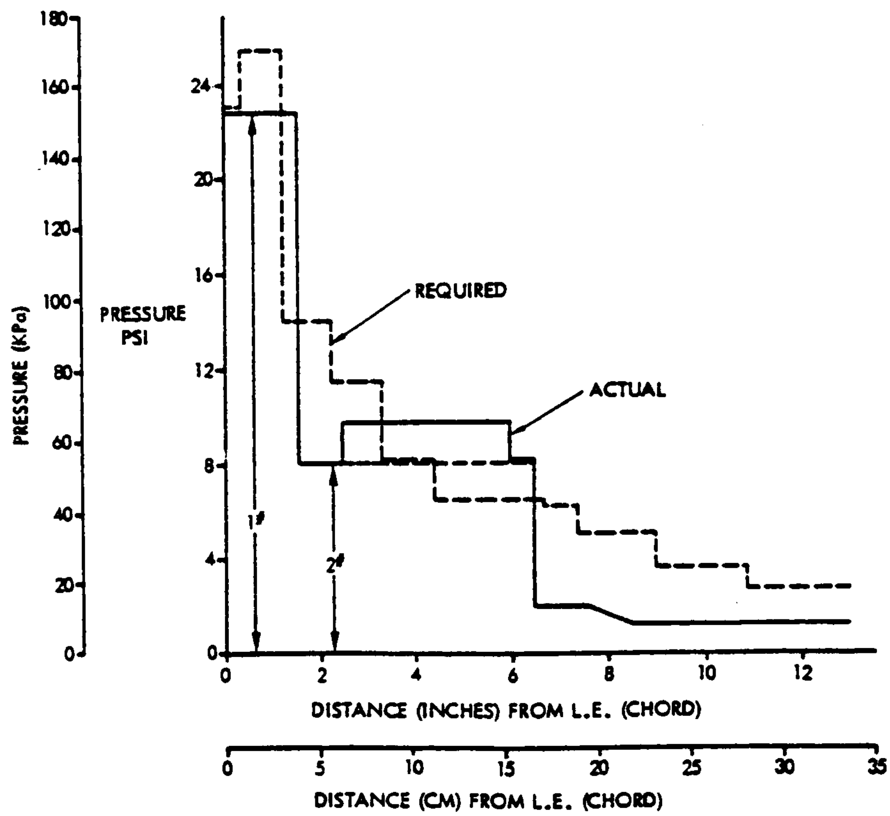


Figure 253. Upper Surface Pressure Distribution at 210% x Limit Load

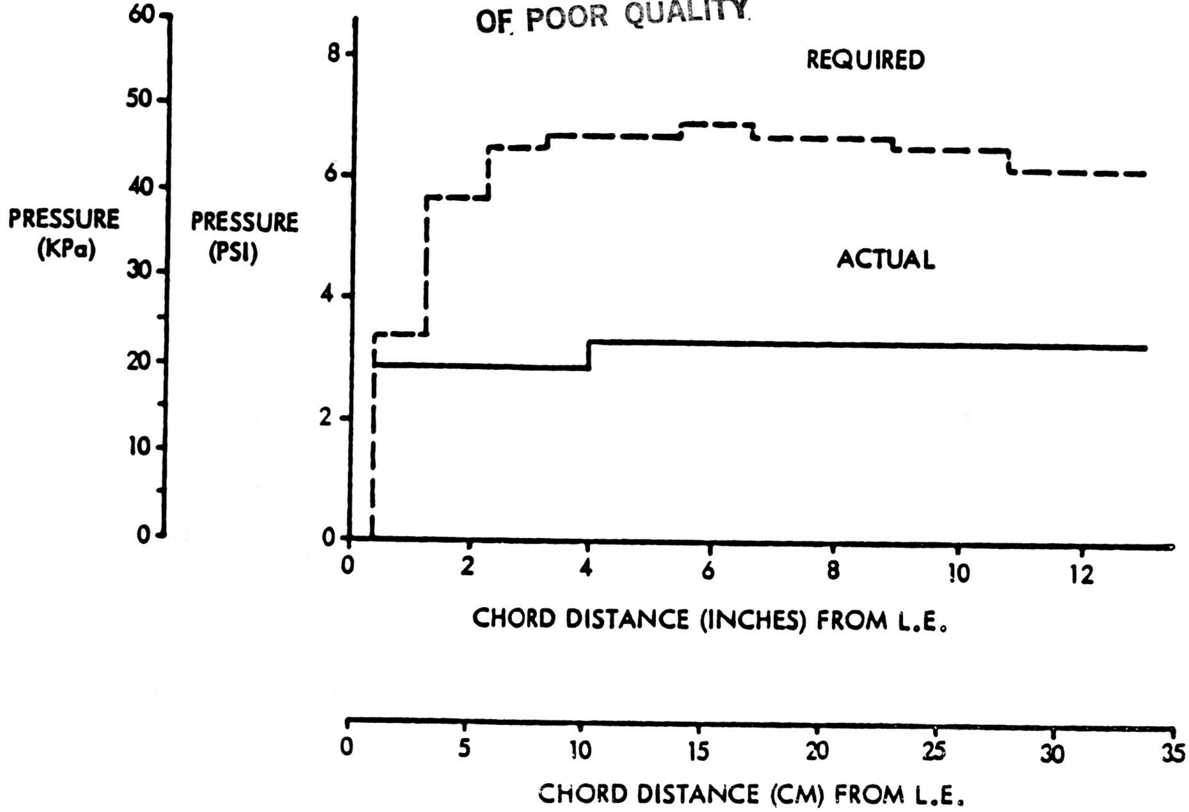


Figure 254. Lower Surface Pressure Distribution at 210% x Limit Load

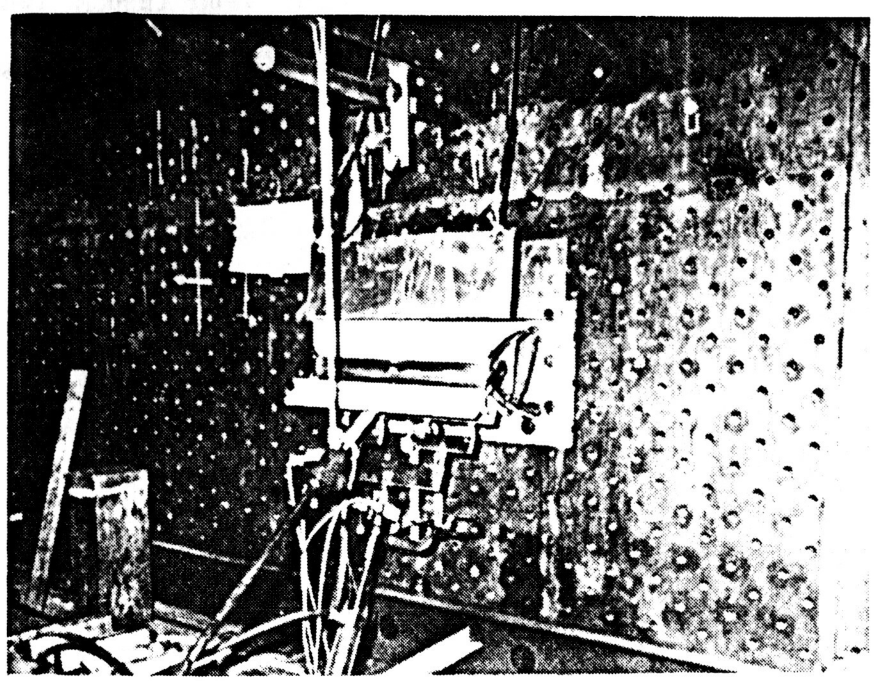


Figure 255. Overall View of Ultimate Test

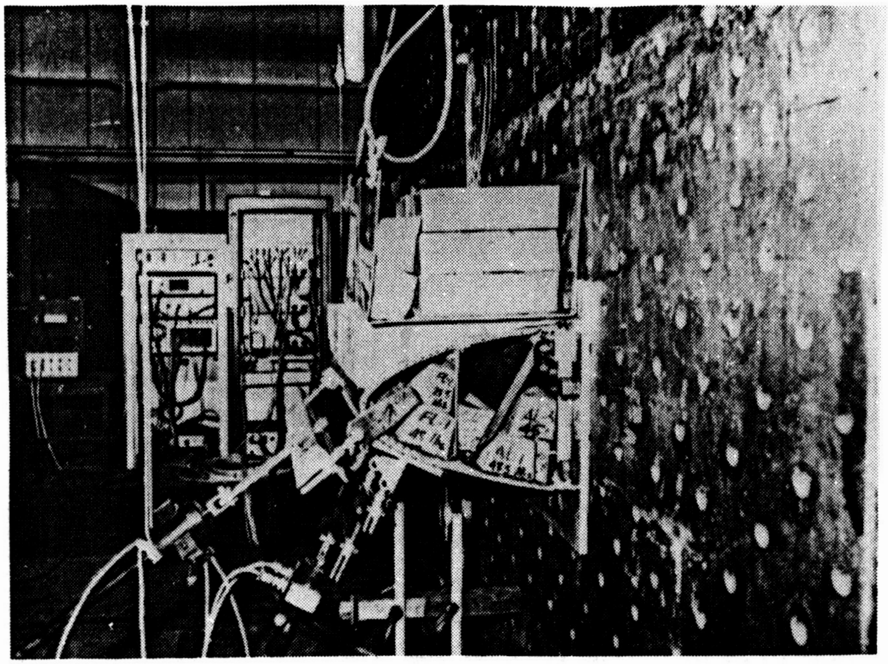


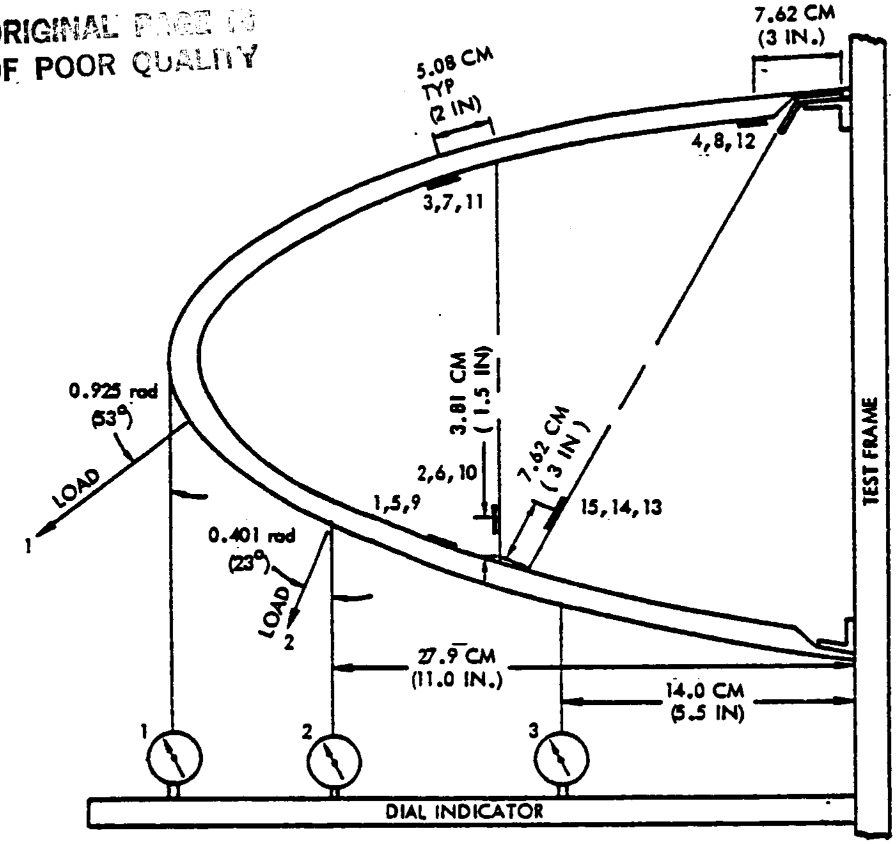
Figure 256. Loading Arrangement Details

9.11.2 SM-6 Summary of Results

Initially, the lower surface was loaded with lead weight to produce the distribution shown in Figure 254 (note this was only to limit load). The upper surface was incrementally loaded by the hydraulic actuators to produce the pressure distribution shown in Figure 253. See Table 35 for jack loads. The strain and deflection data recorded during this loading are shown in Tables 36 and 37. See Table 38 for slot widths; note small change from 40 percent to 100 percent of limit load. In all cases, the total upper surface load is equal to the actuator loads plus linkage weight of 19.1 Kg (42 lb) plus the 182.1 Kg (401.5 lb) internally distributed load.

The test goal was to test the panel to failure. Incremental loading was begun; at 210 percent of limit load (on upper surface) a load tension pad disbonded and testing was stopped. However, no damage to the test article was detected. With the upper surface loaded to 210 percent of limit load and the lower surface loaded to limit, the combined load exceeded 1.5 times limit load on both surfaces; therefore, the test was considered to have exceeded ultimate load when using a factor-of-safety of 1.5.

Acoustic emission (AE) techniques were used to monitor the test article to locate any damage that might occur during loading. Previous work on graphite-epoxy specimens (Reference 11) had shown that resin matrix cracking and fiber fracture can be located if they occur during loading. If no damage occurs, the test article is relatively quiet.



STRAIN GUAGE LOCATION

3.8 CM (1.5 IN) FROM INBOARD END	22.9 CM (9 IN) FROM OUTBOARD END	3.8 CM (1.5 IN) FROM OUTBOARD END
1	5	9
2	6	10
3	7	11
4	8	12
13	14	15

Figure 257. Strain Gauge and Dial Indicator Locations

TABLE 35. APPLIED JACK LOADS

PERCENT LIMIT UPPER SURFACE	LOAD LEVEL (Kg)									PAD + LINKAGE WEIGHT (Kg)
	20	60	100	140	160	180	190	200	210	
LOAD 1	48.1	150.6	255.4	360.6	416.4	468.6	488.5	513.5	543.0	12.7
LOAD 2	12.2	94.3	202.3	308.9	362.4	415.0	451.8	465.8	496.2	6.4

PERCENT LIMIT UPPER SURFACE	LOAD LEVEL (LB)									PAD + LINKAGE WEIGHT (LBS)
	20	60	100	140	160	180	190	200	210	
LOAD 1	106	332	563	795	918	1033	1077	1132	1197	28
LOAD 2	27	208	446	681	799	915	966	1027	1094	14

LOADS WERE APPLIED IN A DIRECTION NORMAL TO THE TANGENT OF THE LE SPECIMEN CURVATURE AT THE LOAD POINT. FOR JACK ANGLES SEE FIGURE 9.11.5

JACK LOADS INCLUDED THE WEIGHT OF PADS AND LINKAGE, BUT DID NOT INCLUDE THE DEAD WEIGHT

TABLE 36. STRAIN DATA

STRAIN GAUGE NO.*	ZERO W/O LEAD	ZERO W/ LEAD	LOAD LEVEL - PERCENT LIMIT - UPPER SURFACE 20	60	100	140	160	180	190	200	210
1	0	-115	-116	-191	-276	-350	-386	-418	-425	-438	-462
2	1	- 46	- 50	- 52	- 53	- 56	- 58	- 60	- 62	- 62	- 62
3	0	65	68	63	54	49	46	44	39	39	40
4	-6	40	55	81	113	147	165	183	188	195	204
5	0	- 64	- 64	-104	-155	-205	-227	-249	-258	-266	-279
6	0	- 75	- 76	- 81	- 84	- 88	- 90	- 93	- 92	- 94	- 93
7	-8	100	107	130	156	184	199	214	224	233	243
8	-1	78	88	93	102	111	117	122	130	133	137
9	-3	- 80	- 92	-141	-202	-261	-290	-318	-334	-344	-362
10	2	- 55	- 62	- 69	- 73	- 81	- 86	- 92	- 97	- 99	-102
11	0	- 84	106	137	176	216	240	262	280	294	307
12	1	125	153	168	194	221	233	243	254	256	259
13	5	204	222	301	388	471	517	562	596	618	645
14	2	227	261	351	465	573	630	683	706	721	750
15	0	190	206	261	334	409	447	483	508	526	550

* FOR LOCATION, SEE FIGURE 9.11.5

TABLE 37. DEFLECTION DATA

LOAD CONDITION PERCENT LIMIT LOAD UPPER SURFACE	CONTOUR DEFLECTION AT DIAL GAUGES * #1		#2		#3		COMMENT
	CM	IN	CM	IN	CM	IN	
ZERO	0	0	0	0	0	0	W/O LEAD DEAD WT
ZERO + WT	0.056	0.022	0.051	0.020	0.018	0.007	W/LEAD DEAD WT
20	0.076	0.030	0.061	0.024	0.013	0.005	
60	0.094	0.037	0.086	0.034	0.025	0.010	
100	0.127	0.050	0.119	0.047	0.030	0.012	
ZERO + WT	0.071	0.028	0.056	0.022	0.020	0.008	W/LEAD DEAD WT
20	0.071	0.028	0.061	0.024	0.020	0.008	
60	0.097	0.038	0.086	0.034	0.025	0.010	
100	0.130	0.051	0.122	0.048	0.030	0.012	
140	0.163	0.064	0.157	0.062	0.036	0.014	
160	0.180	0.071	0.175	0.069	0.041	0.016	
180	0.198	0.078	0.193	0.076	0.043	0.017	
190	0.211	0.083	0.206	0.081	0.046	0.018	
200	0.218	0.086	0.216	0.085	0.048	0.019	
210	0.229	0.090	0.226	0.089	0.051	0.020	

*ALL MOVEMENT IS IN THE DOWN DIRECTION

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TABLE 38. SLOT WIDTHS

LOAD CONDITION PERCENT LIMIT LOAD UPPER SURFACE	SLOT NUMBER			
	U_8		U_9	
	CM	IN	CM	IN
40	0.0046	0.0018	0.0051	0.0020
100	0.0043	0.0017	0.0041	0.0016

The test article was monitored continuously during loading. Some general resin matrix cracking occurred, but the signal level was very low, indicating that microscopic size crazing occurred. No resin matrix microcracking of the type associated with delaminations and splitting parallel to the fiber direction was detected. No fiber fracture was detected.

9.12 SONIC FATIGUE TEST (TEST SM-7)

The objectives of the sonic fatigue test were to (1) determine the resonance modes of the leading-edge test article, (2) measure dynamic strains, (3) verify the sonic fatigue life of the leading-edge section, and (4) verify the durability of the plumbing contained within the leading-edge. All objectives were met.

9.12.1 SM-7 Test Article Description and Instrumentation

The outboard portion, 1.07m (3.50 ft), of the leading-edge assembly was used as the sonic fatigue test article. Twelve uniaxial strain gauges were mounted on the test article to measure typical dynamic strain levels and to sense structural resonances. The instrumented test article was mounted vertically in the Lockheed-Georgia sonic fatigue test facility. Microphones were used to monitor and control the test acoustic spectrum.

9.12.2 Structural Modes and Damping

The test article structural modes and damping data were determined by low level sinusoidal acoustic excitation and are shown in Table 39. Damping values were obtained from strain response plots using the half-power method. Initially, time histories of strain decay were to be used to obtain damping values; however, the decay of the acoustic excitation was slower than the strain decay. Typical graphite structural damping is about 1 percent and typical aluminum structural damping is about 2 percent. Measured damping values shown in Table 39 are consistent with typical values and are acceptable.

9.12.3 Test Procedure and Results

A scatter factor of 2 was used in determining the time and level of all three accelerated-life sonic fatigue tests. The LFC leading-edge test article was exposed for one hour to a 131 dB overall random acoustic level. This was an accelerated-life test to represent the time in cruise at the actual acoustic

TABLE 39. STRUCTURAL MODES AND DAMPING

STRUCTURAL MODE	NATURAL FREQUENCY (Hz)	DAMPING (%)
TITANIUM/GRAPHITE-EPOXY SKIN	415	0.8
GRAPHITE-EPOXY AFT DIAPHRAGM	500	1.0
ALUMINUM FORWARD DIAPHRAGM	245	~ 3.5

level experienced by the JetStar wing leading-edge in cruise. Next, an accelerated-life test to represent the actual time at the takeoff level was simulated by a one-hour random acoustic test at 137dB overall. The last test was conducted at the actual acoustic level experienced by the wing leading-edge during thrust reverse. In this test, the article was subjected to 148.5 dB overall for ten hours.

During these random tests, the acoustic level near the surface of the test article was monitored on-line. The specified excitation levels were not exceeded. Strain response was monitored and measured on-line. The maximum overall strains measured were less than 40 μcm/cm (micro inches per inch), and considerably less than the 350 μcm/cm (micro inches per inch) allowable for this structure. Typical failures in graphite-epoxy structures are evidenced by shifts in modal response frequencies. However, comparisons of strain response plots obtained near the beginning and near the end of the ten-hour test showed no frequency shifts. Therefore, based on this criteria, no failures in the graphite-epoxy structures are indicated by the strain data.

9.12.4 SM-7 Summary of Results

The LFC leading-edge sonic fatigue test section sustained ten hours exposure to the acoustic environment estimated for the reverse thrust case of the JetStar, i.e., 148 dB overall, with the following results:

(a) No visual damage was noted to either the structure (internal and external) or to the plumbing

(b) No shifts were noted in the structural resonances detected by strain gages, which would indicate delamination of the graphite-epoxy structure

(c) All strain levels measured on the test article during testing were very low

(d) Areas of disbonded outer titanium skin determined by tap testing after sonic fatigue tests, coincided with areas of disbonded outer skin noted prior to testing

(e) Pressure tests of the LFC leading-edge plumbing showed virtually the same leak rate before and after sonic fatigue tests.

A pressure leakage check of the plumbing system was made before and after the sonic fatigue tests and the results were compared, Table 40. In both leak tests, a slight amount of leakage was found in the system and determined to occur around some clamped hose connections. However, the amount of leakage after sonic fatigue tests was slightly less than that prior to those tests.

From these results, it is concluded that the JetStar laminar flow control leading-edge assembly will not be structurally or operationally affected or impaired by the intended 1500 hours of operation in the NASA JetStar aircraft acoustic environment.

9.13 SC-3 CLEANING/ANTI-ICING LIQUID PROPERTIES (LAP SHEAR)

Twenty lap shear specimens representing the graphite/epoxy fabric structure to titanium skin bond were fabricated, exposed, and tested. The test specimens were fabricated using FM123-4 film adhesive cured at 93.3°C (200°F). A 4-hour exposure to a 60/40 PGME solution at 48.9°C (120°F) followed by 20 hours of air dry at room temperature was conducted for 10 and 20 cycles. Five specimens were

Number	Test Condition	Test Temperature	Number of Exposure Cycles
(1)	Control	Room Temperature	0
(2)	10 Cycles	Room Temperature	10
(3)	20 Cycles	Room Temperature	20
(4)	Cold 20 Cycles	-53.9°C (-65°F)	20

Specimens geometries, maximum load, maximum computed stress and failure modes are presented in Table 41.

The computed stress for all 20 tested specimens exceed the design allowable of 6,205 KPa (900 psi) with a high margin-of-safety. Therefore, no degradation of the FM123-4 bond is anticipated when exposed to the 60/40 PGME solution.

TABLE 40. LEAK TEST RESULTS-SONIC FATIGUE

SLOT NUMBER	PRE-TEST				POST TEST			
	MAX VAC CM/Hg	MAX VAC IN/Hg	30 SEC CM/Hg	30 SEC IN/Hg	MAX VAC CM/Hg	MAX VAC IN/Hg	30 SEC CM/Hg	30 SEC IN/Hg
U6	2.03	0.80	9.91	3.90	2.54	1.00	9.65	3.80
L1	3.05	1.20	60.96	24.00	3.05	1.20	55.88	22.00
D1	6.86	2.70	-	-	6.35	2.50	-	-
L6	3.81	1.50	67.31	26.50	3.56	1.40	68.58	27.00
U1	5.08	2.00	-	-	5.08	2.00	-	-
D5	5.84	2.30	53.09	20.90	5.59	2.20	58.93	23.20
D6	3.18	1.25	63.50	25.00	2.79	1.10	56.90	22.40

9.14 MANUFACTURING DEVELOPMENT - SLOT CUTTING AND MEASURING DEVELOPMENT (TEST MD-1)

Procedures used in fabricating and measuring the slot widths of the Phase I Wind Tunnel test article described in NASA Contractor Report 159253, September 1980, were not satisfactory. Precision slot cutting procedures were not established and slot width measuring methods were crude, slow, and lacked the accuracy desired. Roll forming the titanium skin also had presented problems. Residual stress between the composite and structure and the titanium skin as a result of curing the adhesive at elevated temperature was also a concern.

A manufacturing development program was developed to address these concerns. Four basic objectives were established:

(a) Develop slot cutting procedures using existing equipment.

(b) Develop adhesive curing procedures to reduce residual stress between the structure and the skin.

(c) Solve titanium forming problems.

(d) Verify that the Wilson Airless Air Gauge will measure slot width effectively.

TABLE 41. LAP SHEAR SPECIMENS

SPECIMEN NUMBER	MAXIMUM LOAD (KG)	MAXIMUM STRESS (KPa X 10^{-3})	JOINT LAP WIDTH (CM)	JOINT LAP LENGTH (CM)	JOINT LAP AREA (CM^2)	FAILURE MODE - % GRAPHITE COHESIVE	FIBERGLASS TO GRAPHITE	ADHESIVE TO GRAPHITE
CONTROL								
1	870.9	25.51	2.565	1.321	3.419	60	40	
2	875.4	25.51	2.565	1.295	3.355	40	60	
3	857.3	24.13	2.565	1.346	3.484	30	70	
4	1,020.6	33.09	2.565	1.321	3.419	85	15	
5	986.6	27.58	2.565	1.372	3.548	90	10	
AVERAGE		(26.20)						
10 CYCLES								
6	873.2	26.20	2.540	1.295	3.290	70	30	
7	848.2	24.82	2.540	1.321	3.355	80	20	
8	920.8	26.20	2.540	1.372	3.484	90	10	
9	886.8	25.51	2.540	1.346	3.419	50	50	
10	410.5	11.72	2.540	1.346	3.419	20	50	30
AVERAGE		(22.75)						
20 CYCLES								
11	979.8	29.65	2.565	1.270	3.226			100
12	850.5	24.82	2.565	1.295	3.355	25	25	50
13	923.1	26.20	2.565	1.346	3.484	90		10
14	914.0	26.20	2.565	1.321	3.419	100		
15	968.4	28.27	2.565	1.295	3.355	100		
AVERAGE		(26.89)						
COLD 20 CYCLES								
16	687.2	19.99	2.565	1.321	3.419	20	80	
17	737.1	21.37	2.565	1.321	3.419	60	40	
18	680.4	19.30	2.565	1.321	3.419	30	70	
19	716.7	20.68	2.540	1.346	3.419	30	70	
20	671.3	19.30	2.540	1.346	3.419	50	50	
AVERAGE		(19.99)						

TABLE 41. LAP SHEAR SPECIMENS (CONT'D)

SPECIMEN NUMBER	MAXIMUM LOAD (LB)	MAXIMUM STRESS (LB/IN^2)	JOINT LAP WIDTH (IN)	JOINT LAP LENGTH (IN)	JOINT LAP AREA (IN^2)	FAILURE MODE (%)		
						GRAPHITE COHESIVE	FIBERGLASS TO GRAPHITE	ADHESIVE TO GRAPHITE
CONTROL								
1	1920	3700	1.01	0.52	0.53	60	40	
2	1930	3700	1.01	0.51	0.52	40	60	
3	1890	3500	1.01	0.53	0.54	30	70	
4	2250	4800	1.01	0.52	0.53	85	15	
5	2175	4000	1.01	0.54	0.55	90	10	
AVERAGE		(3800)						
10 CYCLES								
6	1925	3800	1.00	0.51	0.51	70	30	
7	1870	3600	1.00	0.52	0.52	80	20	
8	2030	3800	1.00	0.54	0.54	90	10	
9	1955	3700	1.00	0.53	0.53	50	50	
10	905	1700	1.00	0.53	0.53	20	50	30
AVERAGE		(3300)						
20 CYCLE (ROOM TEMPERATURE)								
11	2160	4300	1.01	0.50	0.50			100
12	1875	3600	1.01	0.51	0.52	25	25	50
13	2035	3800	1.01	0.53	0.54	90		10
14	2015	3800	1.01	0.52	0.53	100		
15	2135	4100	1.01	0.51	0.52	100		
AVERAGE		(3900)						
COLD 20 CYCLE								
16	1515	2900	1.01	0.52	0.53	20	80	
17	1625	3100	1.01	0.52	0.53	60	40	
18	1500	2800	1.01	0.52	0.53	30	70	
19	1580	3000	1.00	0.53	0.53	30	70	
20	1480	2800	1.00	0.53	0.53	50	50	
AVERAGE		(2900)						

9.14.1 Test Article Description

The wind tunnel model was deskinned by peeling the titanium from the surface. Damaged areas were filled and smoothed as necessary to give a good surface. All old plumbing connections were removed since suction flow requirements for the LEFT program are significantly different from those for which the test section had been originally designed. Where the wind-tunnel test article had a slot duct of 0.381 x 0.203 cm (0.15 x 0.080 in), the flight article slot ducts would be much smaller. A 0.127 cm (0.050 in) radius semi-circular slot duct was chosen for the flight-test article.

Two washing and three suction slots were reconfigured to the new shape. New metering holes were drilled. The new requirements called for a 0.074 cm (0.029 in) diameter metering hole. Significant difficulties were experienced in drilling holes this small with hand-held equipment using the very brittle solid carbide drills.

Figure 258 shows the test article after skin removal while Figure 259 shows a close-up of the reworked slot ducts.

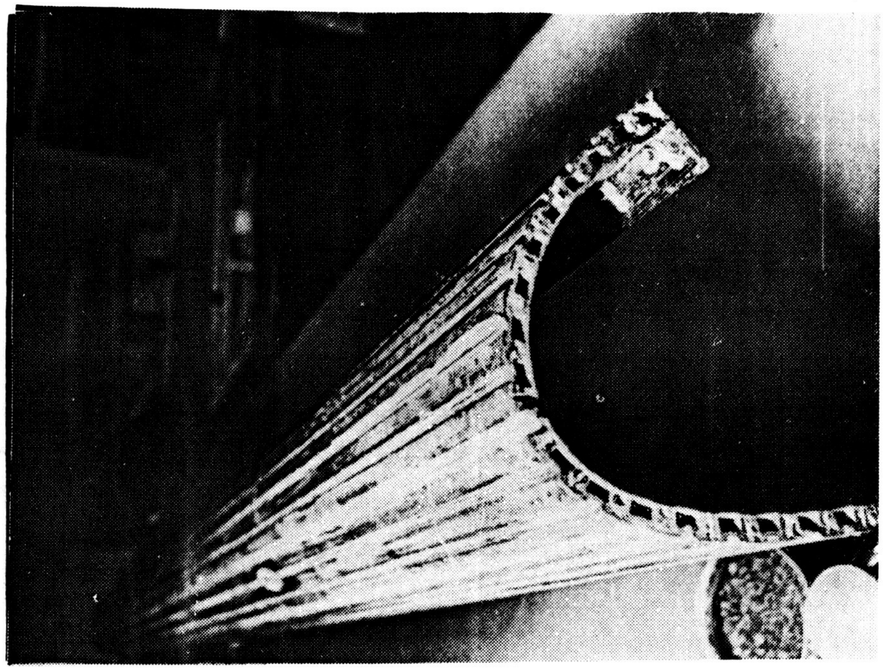

Figure 258. View of Test Article with Old Skin Removed

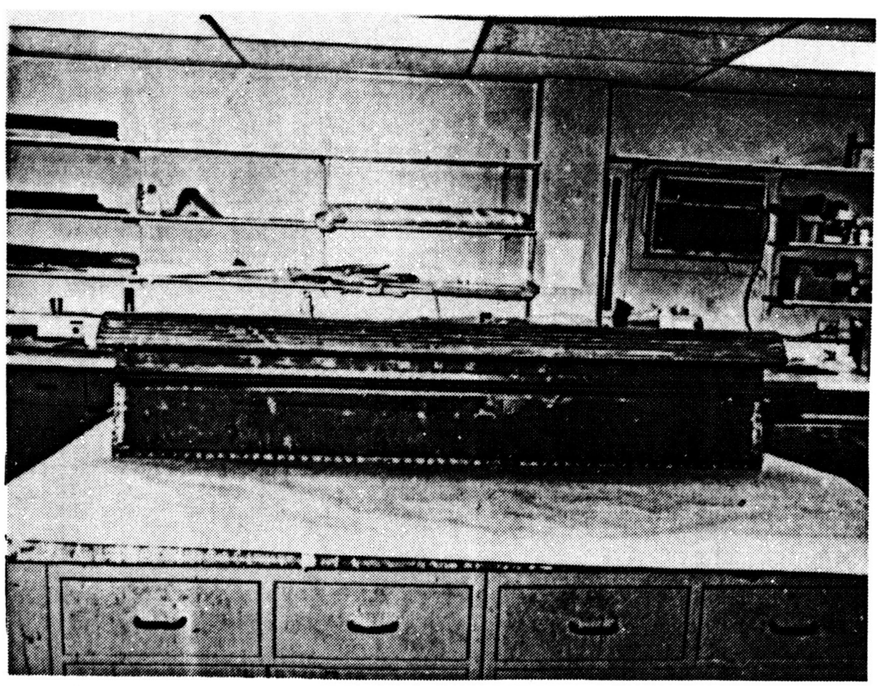

Figure 259. Close-Up View of Reworked Slot Ducts

Half the article was skinned with roll formed 0.041 cm (0.017 in) titanium while the other half was covered with a duplex skin consisting of two layers of 0.020 cm (0.008 in) titanium foil bonded together with Hysol EA9628 adhesive. Based on test results developed in test SM-1, FM 123-4 adhesive, cured for three hours at 93.3°C (200°F), was used to bond the skins to the substructure. The 0.020 cm (0.008 in) titanium foil skins were bonded on the same operation. Bond pressure was 241.3 KPa (35 psi).

9.14.2 Slot Cutting Tests

After bonding, the test article was mounted on a mill for slotting using 0.0076 cm (0.003 in) and 0.0102 cm (0.004 in) high speed steel jewelers saws. These saws were procured from Cleveland Twist Drill Co. The saws were 6.985 cm (2.75 in) in diameter and had 240 teeth. Initial tests showed the saws to be out-of-round with runouts up to 0.0127 cm (0.005 in). Contacts with the manufacturer indicated that this was the closest they could make the saws.

Several coolants compounded for titanium were investigated. Ti cut SRO by Pemco at a 20/1 mixture was found to be the best coolant. It was applied as a mist directly to the cutting face of the saw. Figure 260 shows a jewelers saw mounted in the milling machine with the mist applicator positioned to apply spray correctly. It was necessary to use a climb-cut so that coolant was applied directly to the cutting tooth. A conventional milling cut caused saw failure, probably because the cutting tooth was shielded from the mist and was not properly lubricated.

Results of the cutting tests are shown in Table 42. In general, 10.16 to 12.70 cm (4 to 5 in) per minute feed rate at 68 RPM gave the best results. Saw breakage was probably due to slot movement causing the saw to pinch or wear in the mill table feed.

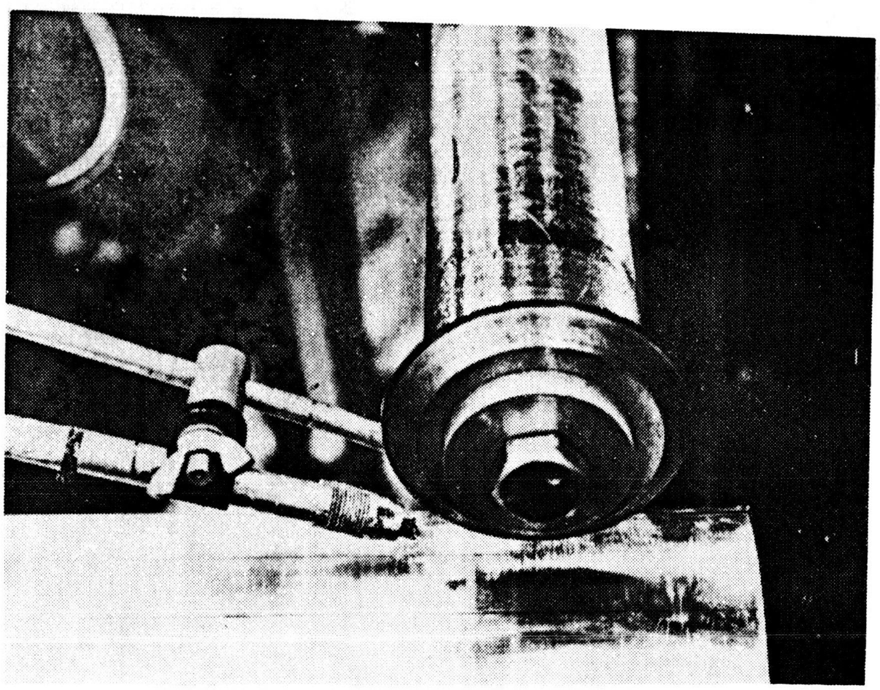

Figure 260. Jewelers Saw Set Up for Cutting Slots

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9.14.3 Results

The five reworked slots were connected to a prototype system for flow tests, as illustrated in Figure 261. Leakage from slot to slot was apparant when the initial tests were made. The probable causes of the leakage were unbonded areas in the skin to structure and internal leakage through porosity in the composite structure.

(a) Slot cutting - Results were quite successful with the biggest improvement being the coolant selector and application methods, however, slot closure was still a problem.

(b) Adhesive curing procedure - It has been hoped that the adhesive could be cured at 82.2°C (180°F) to reduce the residual stress between the substructure and the titanium skin. Test results showed the strength when cured at 82.2°C (180°F) to be low; therefore, a 93.3°C (200°F) cure was established. Residual stress is probably not the cause for the slot closure, rather closure is probably relaxation of the titanium when it is cut.

TABLE 42. SLOT CUTTING TEST RESULTS

SLOT NO.	BLADE RUN-OUT (CM)	SPEED RPM	FEED CMPM	CHIP LOAD/ TOOTH	NO. OF CUTS W/ BLADE	DEPTH FROM SURFACE (CM)	REMARKS
L-1	0.005	68	7.77	.00019	2	0.089	
L-2	0.005	68	7.77	.00019	3	0.089	
L-3	0.013	68	19.35	.00046	1	0.089	LAST 73.7 CM CUT TO L-4 SPECS.
L-4	0.013	68	19.35	.00046	2	0.089	2 TEETH BROKEN
L-5	0.013	68	11.40	.00027	1	0.089	
L-6	0.013	68	11.40	.00027	2	0.089	
L-7	0.008	68	11.40	.00027	1	0.089	0.010 CM BLADE
L-8	0.013	68	11.40	.00027	1	0.089	0.010 CM BLADE
L-9	0.013	68	11.40	.00027	2	0.089	BLADE BROKE COMING OUT OF CUT; 0.010 CM BLADE
L-10	0.010	68	11.40	.00027	1	0.089	0.010 CM BLADE
L-11	0.010	68	11.40	.00027	2	0.089	0.010 CM BLADE
W-5	0.005	68	7.77	.00019	1	0.089	
W-4	0.010	68	11.40	.00027	2	0.089	
W-3	0.009	68	7.77	.00019	1	0.089	+1 RECUT
W-2	0.010	68	11.40	.00027	1	0.089	DENT AT END OF SLOT
W-1	0.013	68	11.40	.00027	1	0.089	
R-1	0.013	68	11.40	.00027	2	0.089	
R-2	0.006	68	11.40	.00027	4	0.076	+1 RECUT @ 0.089 CM DEPTH
R-3	—	—	—	—	-	—	UNSUITABLE SLOT
U-1	0.006	68	11.40	.00027	3	0.076	U-9 BLADE
U-2	0.013	68	11.40	.00027	3	0.089	
U-3	0.006	68	11.40	.00027	2	0.076	U-9 BLADE
U-4	0.013	68	11.40	.00027	4	0.089	
U-5	0.013	68	11.40	.00027	5	0.089	
U-6	0.013	68	11.40	.00027	6	0.089	
U-7	0.013	68	11.40	.00027	7	0.089	
U-8	0.013	68	11.40	.00027	8	0.089	BLADE DULL PUSHING UP BURR
U-9	0.006	68	11.40	.00027	1	0.089	LOWEST RUN-OUT OF BLADES CHECKED THUS FAR.

TABLE 42. SLOT CUTTING TEST RESULTS (CONT'D)

SLOT NO.	BLADE RUN-OUT (IN.)	SPEED RPM	FEED IPM	SHIP LOAD/ TOOTH	NO. OF CUTS W/ BLADE	DEPTH FROM SURFACE (IN.)	REMARKS
L-1	.002	68	3.06	.00019	2	.035	
L-2	.002	68	3.06	.00019	3	.035	
L-3	.005	68	7.62	.00046	1	.035	LAST 29" CUT TO L-4 SPECS.
L-4	.005	68	7.62	.00046	2	.035	2 TEETH BROKEN
L-5	.005	68	4.49	.00027	1	.035	
L-6	.005	68	4.49	.00027	2	.035	
L-7	.003	68	4.49	.00027	1	.035	.004" BLADE
L-8	.005	68	4.49	.00027	1	.035	.004" BLADE
L-9	.005	68	4.49	.00027	2	.035	BLADE BROKE COMING OUT OF CUT; .004" BLADE
L-10	.004	68	4.49	.00027	1	.035	.004" BLADE
L-11	.004	68	4.49	.00027	2	.035	.004" BLADE
W-5	.002	68	3.06	.00019	1	.035	
W-4	.004	68	4.49	.00027	2	.035	
W-3	.0035	68	3.06	.00019	1	.035	+1 RECUT
W-2	.004	68	4.49	.00027	1	.035	DENT AT END OF SLOT
W-1	.005	68	4.49	.00027	1	.035	
R-1	.005	68	4.49	.00027	2	.035	
R-2	.0025	68	4.49	.00027	4	.030	+1 RECUT @ .035" DEPTH
R-3	—	—	—	—	-	—	UNSUITABLE SLOT
U-1	.0025	68	4.49	.00027	3	.030	U-9 BLADE
U-2	.005	68	4.49	.00027	3	.035	
U-3	.0025	68	4.49	.00027	2	.030	U-9 BLADE
U-4	.005	68	4.49	.00027	4	.035	
U-5	.005	68	4.49	.00027	5	.035	
U-6	.005	68	4.49	.00027	6	.035	
U-7	.005	68	4.49	.00027	7	.035	
U-8	.005	68	4.49	.00027	8	.035	BLADE DULL PUSHING UP BURR
U-9	.0025	68	4.49	.00027	1	.035	LOWEST RUN-OUT OF BLADES CHECKED THUS FAR.

(c) Titanium forming requirements - Roll forming has never been successful due to the spring-back problem. This was still apparent on the test article skinned with the rolled sheet. It had been hoped that the laminated titanium skin might exhibit less memory for its original shape. This did not prove to be the condition. However, a major skin forming problem was probably caused by bonding the skins together at the same time as the skin to substructure bond was made. The skins should have been laminated at high pressure in a separate operation prior to bonding to the substructure.

(d) Slot width measurement - The Wilson air gauge proved to be an excellent method of measuring slot width in that it is fast and accurate.

9.15 TOOLING AND MANUFACTURING DEVELOPMENT (TEST MD-2)

The MOD7Q airfoil designed as a best compromise to satisfy both Lockheed and McDonnell Douglas requirements placed new and significant risk on the Lockheed manufacturing plan. Radius of curvature at the nose was much tighter

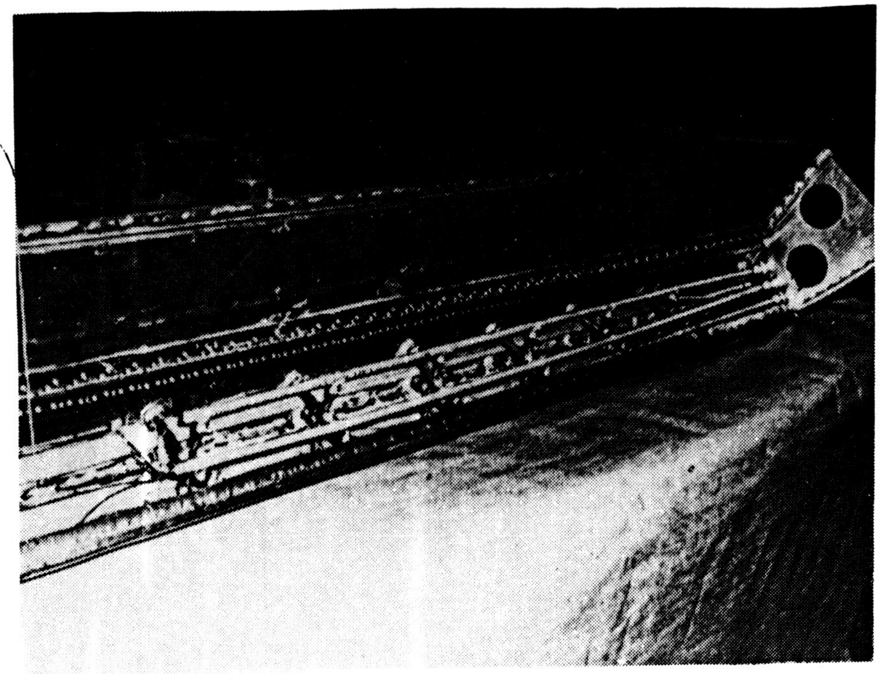

Figure 261. Prototype Plumbing System Installed for Flow Tests

than anything Lockheed had worked with in the past. Smaller and closer spaced slot ducts were required to accommodate the required pressure distribution over the test section. The combination of these changes required a revision of the tooling and manufacturing concepts to new concepts which were untried. With little past experience available to backup these new concepts, it was necessary to conduct a subscale test to validate the planned approach.

Objectives of the subscale test were threefold:

(a) Develop a process for forming the titanium to the new radius with a minimum of tooling.

(b) Develop a process to form 0.0102 to 0.1270 cm (0.040 to 0.050 in) radius slot ducts.

(c) Develop a tooling concept that can be used to produce an LFC leading edge with the small slot ducts.

9.15.1 Test Article Tooling

A test specimen consisting of a 60.96 cm (24 in) section of the MOD7Q leading edge from leading edge station 159.121 to leading edge station 183.121 was selected. Tooling consisted of a conventional fiberglass/epoxy female tool made by the following process:

(a) Templates were cut at 15.24 cm (6 in) stations.

(b) Plaster was swept to the templates.

(c) A plaster female splash was made of the shape.

(d) Cast a plastic-on-plastic plug.

(e) Layup and cure a fiberglass/epoxy tool over the plug.

Since this test was primarily to check out manufacturing techniques, no special precautions were used in making the tool. The station cuts were plotted from loft data on the Gerber plotter. Templates were shot from the "mylars" and filed to shape. It was interesting to note that normal plaster master model tooling practices produced a tool that seemed to meet smoothness requirements.

Slot duct insert - A slot duct of 0.0102 cm (0.040 in) radius was initially planned. The radius was changed to 0.152 cm (0.060 in) due to test results. Compression molding appeared to be the most practical approach to producing these inserts. Fiberglass prepreg was laid up in the mold and cured in an autoclave.

Collector ducts - These ducts were purchased from Morrison Molded Fiberglass Products and were produced by pultrusion.

One of the most critical details of the manufacturing process was the location of each slot duct insert in position within ± 0.0127 cm (± 0.005 in). This was accomplished by developing the surface flat pattern from loft data. Grooves representing the inner contour of the slot duct inserts was then machined into an aluminum tooling plate centered at each slot position. Figure 262 shows the slot duct locator tools.

9.15.2 Test Article Fabrication

The first step in the fabrication process was to form the outer skin subassembly. Slot duct inserts which had been previously molded from fiberglass were positioned in the locator tool. A layer of nylon fabric impregnated with resin was laid up over the inserts, followed by a layer of style 120 fiberglass prepreg. This subassembly was then cured which effectively locked the inserts

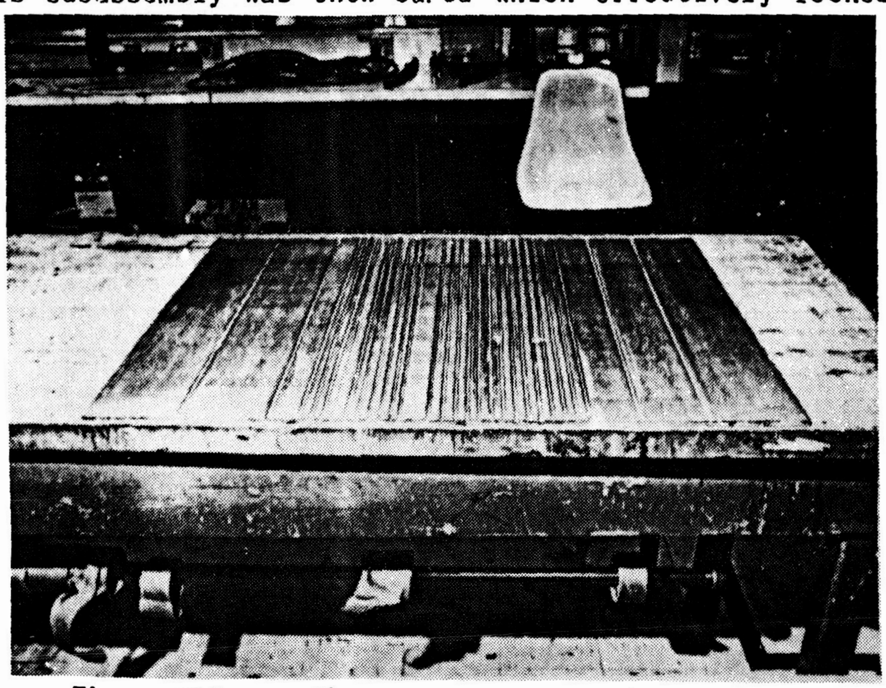

Figure 262. Slot Duct Locator Tool for MD-2

into position. Since the nylon/fiberglass layer can be peeled from the substructure, this item is called the peel ply subassembly and is shown in Figure 263. After removal from the tool, a layer of high tack adhesive was applied to the entire inner surface to bond the graphite/epoxy fabric prepreg to the inserts. A high tack adhesive is needed to hold the fabric in place. It was found that even with a 0.0203 cm (0.008 in) fabric, filler plies were needed between the slot duct inserts to prevent bridging. After the first ply of graphite had been laid up to the peel ply subassembly in the flat, it was transferred to the leading edge tool where the remaining plies were positioned. Cure of the outer skin was accomplished under 586.0 KPa (85 psi) and 176.7°C (350°F). Following this operation the collector ducts were bonded into position with Hysol EA934 adhesive. Following bonding of the collector ducts, the honeycomb core was bonded in place. Figure 264 shows the outer subassembly. The subassembly can be inspected at this point and any problems corrected. Layup and cure of the inner skin followed.

At this point the nylon peel ply is removed from the outer surface. Some early problems with resin flow into the slot duct were solved by placing Teflon coated wire in the duct to prevent it filling with resin. Figure 265 shows the leading edge substructure with part of the peel ply removed. In general a good surface was obtained. Metering holes were drilled by hand which is not practical since 7900 metering holes are required for the LETA. Automatic equipment was used for the flight-test article. Figure 266 is a closeup view of the surface after drilling the metering holes. Figure 267 shows the test article ready for skin bond.

The most attractive titanium skin forming process appeared to be form/bonding two layers of 0.020 cm (0.008 in) foil together. While this was not completely successful in the MD-1 test, the skin forming failure was primarily attributed to the single stage bond at low pressure. Bonding of the skins was to be accomplished at 689.5 KPa (100 psi) using an adhesive specifically formulated for wide area metal-to-metal bonds. While the first try had some voids at

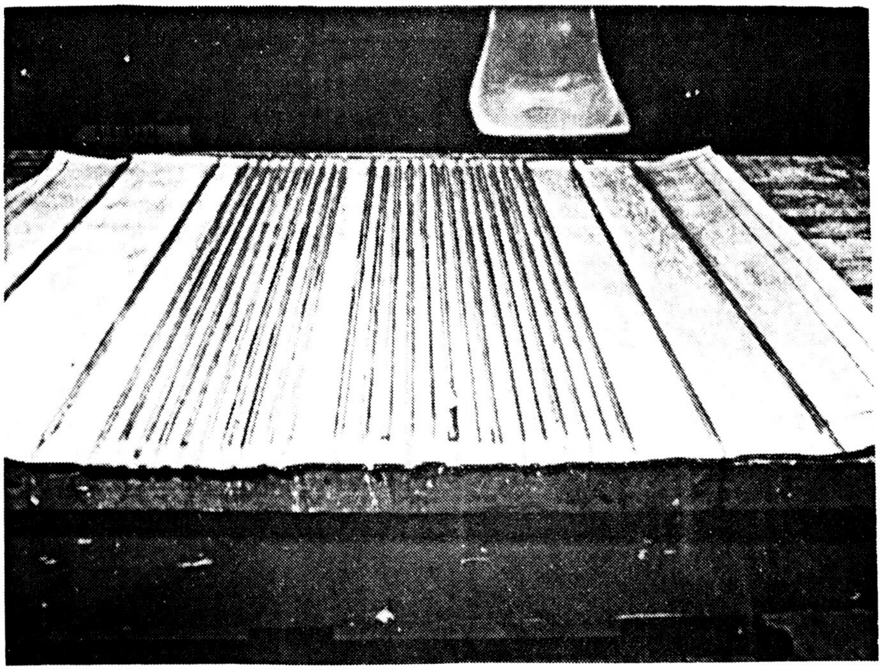

Figure 263. Peel Ply Subassembly

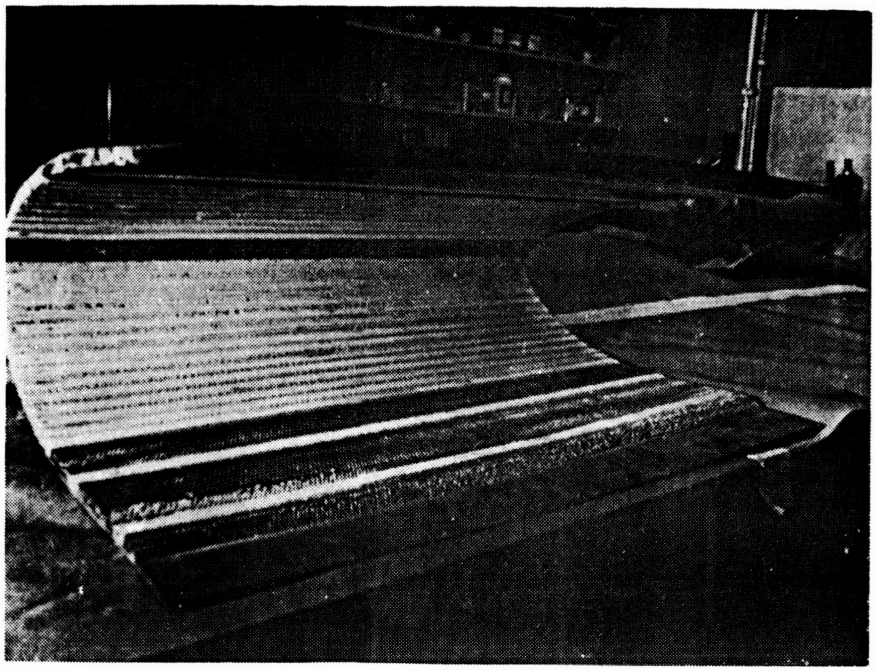

Figure 264. Outer Subassembly with Ducts and Core in Place

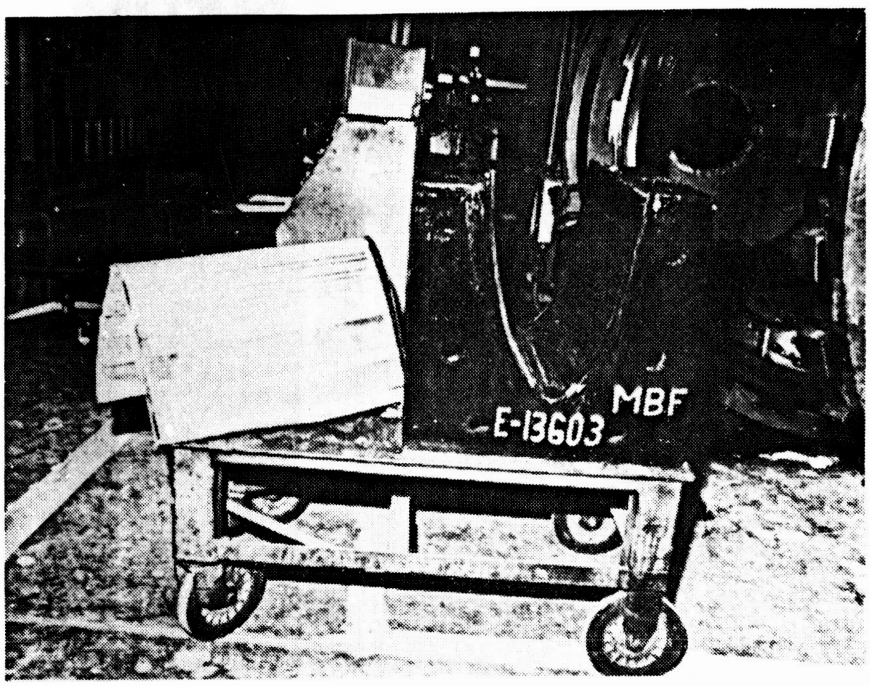

Figure 265. Leading Edge Substructure and Mold. Peel Ply Partly Removed

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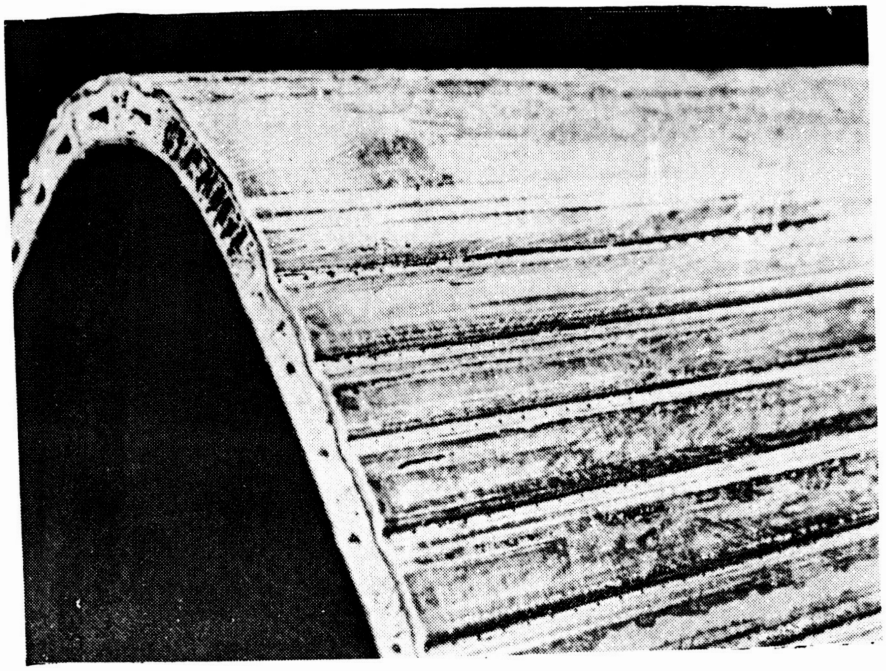

Figure 266. Close-Up View of MD2 Test Article after Metering Holes had been Drilled

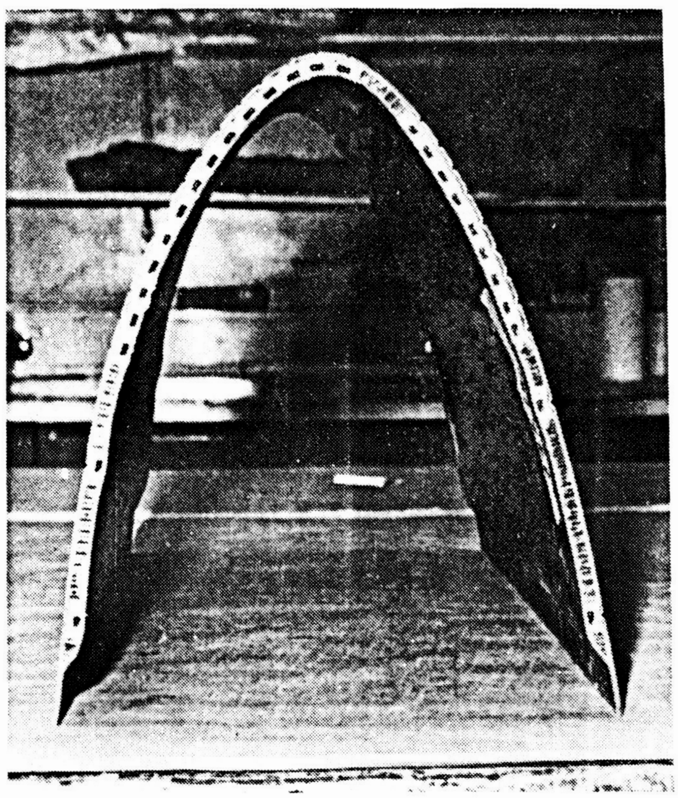

Figure 267. MD-2 Test Article Ready for Skin Bond

the nose, modifications to the process produced an excellent skin assembly the second time.

Bonding of the skin to the substructure was accomplished with the same adhesive used on the MD-1 test article, American Cyanamid FM123-4, which has a very low flow characteristic. Slotting was accomplished using the procedures developed in test MD-1. Slot closure was not a problem. Following skin slotting, the slot widths were measured with the Wilson air gauge. Results are shown in Table 43. A slight delamination at the inboard end was apparent after slotting, but no particular reason was apparent except for possible contamination of the titanium skin prior to bonding. No evidence of slot closure was apparent during slot measurements. The slight burr left from the saw was removed by sanding which produced a very smooth leading edge surface.

9.15.3 MD-2 Test Results

MD-2 was a successful test. This test resolved several manufacturing process uncertainties; therefore, it seemed that an LFC flight test article could be fabricated using the processes developed during this test.

In summary, the major achievements or concerns resolved were:

(a) Small slot ducts can be precision formed by compression molding a fiberglass insert.

(b) Slot ducts can be located with good accuracy by a rather straightforward tooling/fabrication procedure.

(c) The 0.020 cm (0.008 in) graphite/epoxy fabric prepreg will form the skin over the slot-duct inserts. However, filler plies will be required to prevent voids on the outer surface.

(d) Collector ducts can be made from a fiberglass pultrusion.

(e) Two layers of 0.020 cm (0.008 in) titanium foil can be form/bonded into a titanium outer skin.

(f) The leading-edge sandwich structure is stable. No movement was apparent when it was removed from the tool.

(g) Bonding the form/bonded titanium skin to the sandwich did not change the shape of the structure.

(h) No evidence of slot closure was detected while cutting slots.

TABLE 43. MD-2 SLOT WIDTHS

(CM X 10^{-4} - AT APPROXIMATE LES)

SLOT NUMBER	AFTER SLOTTING					SPAR REMOVED - CLEANED					SPAR REPLACED				
	160	165	171	177	181	160	165	171	177	181	160	165	171	177	181
U11	51	64	64	64	64	53	71	76	76	58	76	81	86	76	79
U10	81	89	94	89	89	64	64	71	76	69	76	69	76	89	86
U9	76	89	91	89	81	76	71	76	71	71	89	86	89	81	86
U8	79	79	76	81	81	79	76	76	76	81	86	84	81	76	86
U7	81	81	89	97	86	76	79	81	81	81	79	84	84	84	84
U6	81	81	89	86	89	81	76	76	79	76	81	81	84	91	89
U5	91	94	91	97	97	91	94	91	91	89	99	91	84	91	94
U4	94	89	89	91	97	91	79	86	86	89	91	76	89	97	97
U3	91	84	71	99	97	84	89	81	79	79	89	91	91	91	89
U2	102	91	97	97	91	76	91	97	89	89	97	99	107	99	97
U1	102	97	86	91	97	89	89	89	71	71	102	102	102	102	86
W1	-	81	89	86	81	-	71	76	76	64	-	81	89	84	89
W2	-	76	89	91	84	66	69	74	74	76	79	76	76	81	86
W3	94	99	94	84	97	64	76	81	81	81	91	91	97	97	81
W4	102	104	97	102	97	79	89	89	89	81	102	94	97	97	91
W5	97	86	89	91	89	91	89	84	84	86	102	109	91	81	76
W6	64	86	102	107	91	76	89	89	94	102	84	86	104	104	104
W7	91	86	102	91	89	89	91	89	81	89	91	91	99	94	91
L1	91	94	86	97	97	94	97	89	97	97	102	102	99	102	102
L2	91	91	89	81	79	84	84	89	84	89	91	99	89	94	84
L3	71	81	79	97	81	76	76	74	76	76	79	81	76	84	81
L4	79	79	79	86	89	76	74	76	76	76	81	81	79	81	81
L5	53	74	71	76	79	76	71	71	76	74	76	79	76	79	81
L6	74	76	74	71	76	76	76	69	76	76	79	76	76	79	76
L7	66	76	64	79	76	71	79	79	79	76	76	76	79	81	79
L8	71	64	71	76	76	66	71	76	76	79	76	76	76	76	76

MEASURED WITH WILSON AIR GAUGE

TABLE 43. MD-2 SLOT WIDTHS (CONT'D)

(INCHES X 10^{-4} - AT APPROXIMATE LES)

SLOT NUMBER	AFTER SLOTTING					SPAR REMOVED - CLEANED					SPAR REPLACED				
	160	165	171	177	181	160	165	171	177	181	160	165	171	177	181
U11	20	25	25	25	25	21	28	30	30	23	30	32	34	30	31
U10	32	35	37	35	35	25	25	28	30	27	30	27	30	35	34
U9	30	35	36	35	32	30	28	30	28	28	35	34	35	32	34
U8	31	31	30	32	32	31	30	30	30	32	34	33	32	30	34
U7	32	32	35	38	34	30	31	32	32	32	31	33	33	33	33
U6	32	32	35	34	35	32	30	30	31	30	32	32	33	36	35
U5	36	37	36	38	38	36	37	36	36	35	39	36	33	36	37
U4	37	35	35	36	38	36	31	34	34	35	36	30	35	38	38
U3	36	33	28	39	38	33	35	32	31	31	35	36	36	36	35
U2	40	36	38	38	36	30	36	38	35	35	38	39	42	39	38
U1	40	38	34	36	38	35	35	35	28	28	40	40	40	40	34
W1		32	35	34	32		28	30	30	25		32	35	33	35
W2		30	35	36	33	26	27	29	29	30	31	30	30	32	34
W3	37	39	37	33	38	25	30	32	32	32	36	36	38	38	32
W4	40	41	38	40	38	31	35	35	35	32	40	37	38	38	36
W5	38	34	35	36	35	36	35	33	33	34	40	43	36	32	30
W6	25	34	40	42	36	30	35	35	37	40	33	34	41	41	41
W7	36	34	40	36	35	35	36	35	32	35	36	36	39	37	36
L1	36	37	34	38	38	37	38	35	38	38	40	40	39	40	40
L2	36	36	35	32	31	33	33	35	33	35	36	39	35	37	33
L3	28	32	31	38	32	30	30	29	30	30	31	32	30	33	32
L4	31	31	31	34	35	30	29	30	30	30	32	32	31	32	32
L5	21	29	28	30	31	30	28	28	30	29	30	31	30	31	32
L6	29	30	29	28	30	30	30	27	30	30	31	30	30	31	30
L7	26	30	25	31	30	28	31	31	31	30	30	30	31	32	31
L8	28	25	28	30	30	26	28	30	30	31	30	30	30	30	30

MEASURED WITH WILSON AIR GAUGE

10.0 RECOMMENDATIONS AND CONCLUDING REMARKS

At the time this final report is being published, November 1983, the modifications to incorporate the Lockheed and McDonnell Douglas leading-edge flight test articles are complete except for connection of the LFC systems and instrumentation. NASA Dryden has scheduled the JetStar flight acceptance phase for December 1983. The JetStar LFC systems acceptance is scheduled for March 1984 at which time the aircraft will commence a NASA conducted LFC flight test program continuing until mid-1985. Although the LFC leading-edge flight test program phase is just beginning, some recommendations and concluding remarks are appropriate now.

10.1 FABRICATION RECOMMENDATIONS

As a result of the problems and lessons learned during the fabrication of the flight-test article, the following recommendations on design and process changes are made:

(1) Any "as molded" composite thickness dimensions should be established at the upper limit of the per ply thickness. The upper and lower attachment edge of the LFC-18-3 bond assembly shows a 0.305 cm (0.120 in) thickness dimension, which was based on idealized ply thickness. In practice the assembly molds to a minimum thickness of 0.381 cm (0.150 in). Since location of the lower surface spar cap extension was based on the attachment lip being 0.305 cm (0.120 in) the leading edge was located 0.076 cm (0.030 in) too low. A thickness of at least 0.4406 cm (0.160 in) should be planned and the part then shimmed upon installation.

(2) Graphite/epoxy fabric skins should require splices in the nose region. While not specified, it was desired that the 0.785 rad (45 deg) pieces of fabric not be spliced in the spanwise direction during layup. Such splices, in addition to the ones in the chordwise direction would have increased the thickness of the skin and encroached on the inside envelope. It was extremely difficult to layup the outer skin so that bridging did not occur in the nose region. The tool try article skin formed reasonably well into the nose, however, this was not the case with the flight test article outer skin which required considerable filling to eliminate voids. No piece of fabric should be laid down in a deep female tool without a splice in the apex to allow movement as the material compresses during cure. This splicing processs would insure a dense, void-free skin that conforms exactly to the tool.

(3) The use of premolded slot duct inserts was chosen since it gave a precision slot duct shape and could be located by use of the "peel ply subassembly." In practice this process was not completely successful because:

- o Molding the inserts was labor intensive
- o Slippage could occur on the subassembly during subsequent operations
- o It was almost impossible to get a dense outer skin even with fillers between the inserts to prevent bridging of the fabric.

A better approach now appears to layup and cure a solid composite outer skin of sufficient thickness to allow the slot ducts to be machined into the skin. This would reduce layup time and allow a dense, void-free skin to be obtained. Collector ducts could then be positioned by end location jigs. After the sandwich subassembly is complete, the slot ducts would be machined followed by drilling the metering holes. Also eliminated would be the potential leak paths between metering holes caused by porosity.

(4) A separate adhesive specification should be prepared to control the flow of the adhesive and several samples of the batch tested to ensure that the adhesive properties are as specified. American Cyanamid FM123-4 still appears the best candidate for use. A flow limit can easily be established to assure that a flow condition as experienced on the flight test article would not be repeated.

(5) Forming procedures for titanium also need additional development. Hot-forming in the superplastic region still appears the best approach to obtain a stress-free formed shape. Castable silica also still appears as the best tool material. The only change over the process tried earlier would be to heat the tool and titanium in a furnace so that the skin would be formed at a uniform temperature. Drape forming by using a stainless steel vacuum bag would probably be the preferred procedure.

(6) The use of fiberglass/epoxy tools is not recommended for use in forming an LFC leading edge. A machined steel leading-edge mold would be extremely expensive and difficult to produce. An autoclave cured graphite/epoxy tool properly designed might be a reasonable approach, except producing a master model by splining plaster to templates will always have problems in achieving the tolerances required. A possible approach to producing a lower cost, but more stable tool, would be to cut a master model by a computer controlled three or five axis milling machine. A shell of nickel would then be produced by electroforming over the male model. The electroformed mold would then be stabilized by attaching to a steel frame. Accuracies obtainable by this process are not known but they would be better than those produced by the process used on this program and would be much less expensive.

10.2 CONCLUDING REMARKS

(1) Delivery of the leading-edge flight test article is a major achievement in this project. Lockheed had several unforeseen problems in fabrication which were successfully resolved.

(2) Installation of flight test articles on the JetStar aircraft has been accomplished and has resulted in a fully satisfactory smooth and clean configuration for testing.

(3) Lockheed believes the total aircraft and leading-edge flight test articles should provide the anticipated data base in the forthcoming airline simulation flight test program.

(4) The data base developed in the airline simulation flight test program should allow industry to resolve issues and decisions regarding the

maintainability, reliability, and affordability of LFC systems for future aircraft applications.

(5) The data base developed in the project should aid in the definition of the next initiative and new start for LFC development.

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16. Abstract
Reduction of skin friction drag by suction of boundary layer air to maintain laminar flow has been known since Prandtl's published work in 1904. The dramatic increases in fuel costs and the potential for periods of limited fuel availability provided the impetus to explore technologies to reduce transport aircraft fuel consumption. NASA sponsored the Aircraft Energy Efficiency (ACEE) program beginning in 1976 to develop technologies to improve fuel efficiency. This report documents the Lockheed-Georgia Company accomplishments in designing and fabricating a leading-edge flight test article incorporating boundary layer suction slots to be flown by NASA on their modified JetStar aircraft. Lockheed-Georgia Company performed as the integration contractor to design the JetStar aircraft modification to accept both a Lockheed and a McDonnell Douglas flight test article. McDonnell Douglas uses a porous skin concept for their flight test article. The report describes aerodynamic analyses, fabrication techniques, JetStar modifications, instrumentation requirements, and structural analyses and testing for the Lockheed test article. NASA will flight test the two LFC leading-edge test articles in a simulated commercial environment over a 6 to 8 month period in 1984. The objective of the flight test program is to evaluate the effectiveness of LFC leading-edge systems in reducing skin friction drag and consequently improving fuel efficiency.

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REFERENCES

1. Ballhaus, W. F., Bailey, F. R., and Frick, J., "Improved Computational Treatment of Transonic Flow About Swept Wings," NASA CP-2001, November 1976.

2. Hess, J. L., "Calculation of Potential Flow About Arbitrary Three-Dimensional Lifting Bodies," Navy Technical Report MDC J5679-01, October 1972.

3. "Test of the LFC JetStar Leading Edge Glove Configuration in the Calspan Eight Foot Transonic Wind Tunnel, "Lockheed-Georgia Company Engineering Report LG81ER0072, DRL-I-020, January 1981.

4. Kline, J., "Wind Tunnel Test of a 0.10 Scale Model of the LFC JetStar Leading Edge Glove Configuration," CALSPAN Report No. AA-4007-W-13, April 1981.

5. Peters, Spence E., Jr., "LFC JetStar Stability and Control Report," Lockheed-Georgia Company, April 1981.

6. Cebeci, T., and Kaups, K., "Compressible Laminar Boundary Layer with Suction on Swept and Tapered Wings," Journal of Aircraft, Vol. 14, No. 7, July 1977, pgs 661-667.

7. Srokowski, Andrew J. and Orszag, Steven A., "Mass Flow Requirements for LFC Wing Design," Presented as paper #77-12222, AIAA Aircraft Systems and Technology Meetings, August 22-24, 1977, Seattle, Washington.

8. Brandt, L. B., "Airplane Operational Data Report, "Lockheed-Georgia Company Engineering Report LG81ER0191, July 1981.

9. JetStar Aerodynamics Group, "C-140 Aerodynamics Substantiating Data Report," Lockheed-Georgia Company Engineering Report LG62ER5862, August 1962.

10. Military Performance Group, "C-140 Aerodynamics Substantiating Data Report," Lockheed-Georgia Company Engineering Report LG68ER5862, Revised October 1968.

11. C. D. Bailey, et al, "Acoustic Emission of Impact-Damaged Graphite-Epoxy Composites," Materials Evaluation, May 1979.